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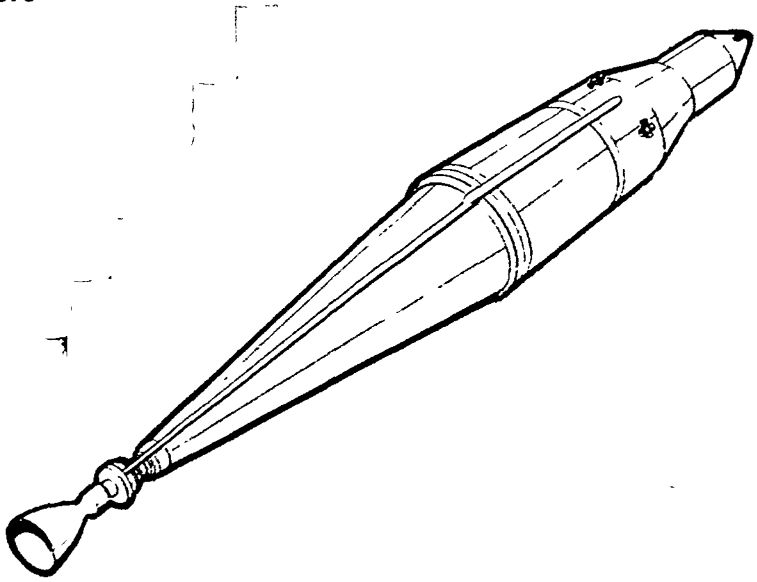
NUCLEAR FLIGHT SYSTEM DEFINITION STUDY

PHASE III FINAL REPORT

PART B.
BASELINE SYSTEM DEFINITION

PREPARED FOR
GEORGE C. MARSHALL
SPACE FLIGHT CENTER

CONTRACT NO. NAS8-24975
SD71-466-3



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SD 71-466-3

NUCLEAR FLIGHT SYSTEM DEFINITION STUDY

PHASE III FINAL REPORT

Volume II - Concept and Feasibility Analysis

Part B. Baseline System Definition

Prepared For

George C. Marshall Space Flight Center
National Aeronautics and Space Administration
under Contract NAS8-24975

April 1971

Prepared By

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II

FOREWORD

The final report on the Phase III Reusable Nuclear Shuttle (RNS) study was prepared by the North American Rockwell Corporation through its Space Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Appendix A of contract NAS8-24975. The contract directed a study of mission requirements, design concepts and definition, performance, operations, facilities, and development activities for the RNS with associated funding and scheduling requirements.

This report is submitted in six volumes with Volume II consisting of three separate books:

I.	(SD 71-466-1)	Executive Summary
II.		Concept and Feasibility Analysis
A.	(SD 71-466-2)	System Evaluation and Capability
B.	(SD 71-466-3)	Baseline System Definition
C.	(SD 71-466-4)	System Engineering Documentation
III.	(SD 71-466-5)	Program Support Requirements
IV.	(SD 71-466-6)	Cost Data (Limited Distribution)
V.	(SD 71-466-7)	Schedules, Milestones, and Networks
VI.	(SD 71-466-8)	Reliability and Safety Analysis

This book of Volume II presents the detail design of the baseline RNS configuration including engineering drawings, subsystems, schematics, componenets, mass properties, and interface requirements. Also presented is a brief description of each of the selected subsystems and the Design Criteria and Constraints.

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IV

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C. F. DeBarro	Configuration Development and Structures
L. S. Nogawski	Loads and Dynamics
J. P. Grocetto	Loads and Dynamics
J. Hardy	Loads and Dynamics
E. Long	Loads and Dynamics
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VI

CONTENTS

CONTENTS (CONTINUED)

ILLUSTRATIONS

ILLUSTRATIONS (CONTINUED)

TABLES

5.0 GENERAL ARRANGEMENT

The recommended RNS baseline resulting from the tradeoffs and detail investigations presented in Volume II-A, Section 4.0, is a single tank conical aft bulkhead configuration with an 8-degree half cone angle and a 25-inch cap radius, launched inverted without the NERVA engine on an INT-21 booster. A nose cone shroud is used over the conical aft bulkhead during the boost phase. The NERVA engine is delivered to orbit via the Space Shuttle and mated to the stage in earth orbit to complete the operational vehicle. The overall dimensions of the vehicle for 300,000 pounds of LH_2 with 5 percent ullage volume are given in Figure 5-1. The stage overall length including the engine is 194 feet. The tank excluding the astrionics bay and thrust structure is approximately 153 feet in length, of which 105 feet is the conical aft bulkhead.

The tank cylindrical section is 33 feet in diameter. The forward bulkhead geometry is an oblate spheroid of aspect ratio 1.5, i.e., the minor half axis is equal to 132.0 inches.

The high fineness ratio conical bulkhead enhances radiation attenuation by combining the benefits of increased separation distance of payload from NERVA and of reduced neutron/gamma energy transmitted into the tank due to the smaller view angle. The configuration meets the radiation dose criterion of 10 REM at the tank top with an external shield weight of 4050 pounds.

HARDWARE TREE

The manufacturing and assembly of the RNS require the development of a fabrication time line and scheduling discussed in Volume III, Section 2.0, and also in Volume V. However, to generate the latter, an identification of the hardware components and their elements is necessary. To this end a Hardware Tree consistent with a Phase A context was generated and is presented in Table 5-1. It identifies the seven major subsystems and their constituent elements. The propellant tank and propellant management are shown further subdivided into their sub-elements.

The subdivision identified under the RNS Hardware Tree is used to guide the description of the inboard profile of the recommended configuration and explain the schematics and block diagrams of the various subsystems. The inboard profile is presented in Figure 5-2. Repeated reference to this figure will be made throughout this section.

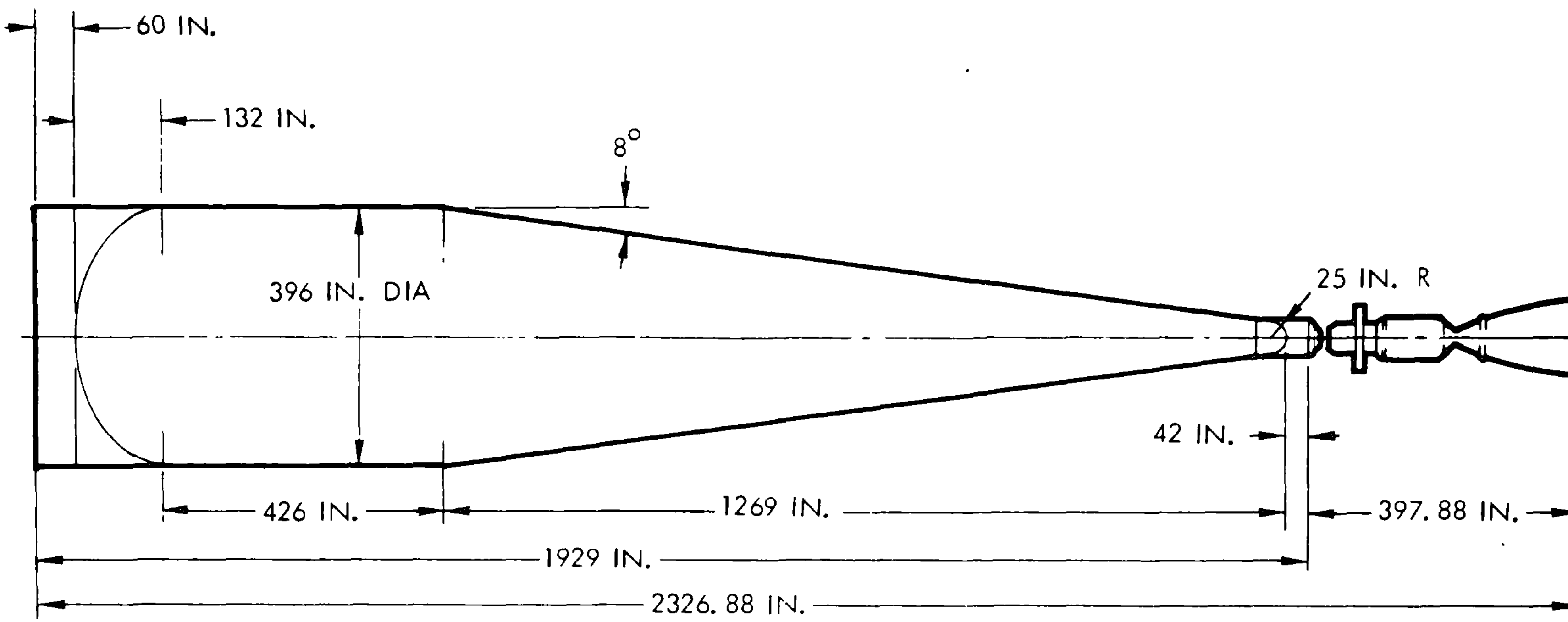

Figure 5-1. Single Tank Baseline Configuration

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Table 5-1. Hardware Tree for Single Tank Conical Aft Bulkhead RNS

REUSABLE NUCLEAR SHUTTLE VEHICLE

- **STRUCTURE**
  - Propellant Tank
    - .Forward Bulkhead
    - .Cylinder Section
    - .Aft Bulkhead
  - Engine Thrust Structure
  - Forward Skirt
  - Aft Skirt
  - Tunnel & Fairing
  - Exterior Finish & Sealer
  - Equipment Support Structure
  - Astrionics Module Structure
  - Aero Nose Cone Structure (Expendable)
- **THERMAL/METEOROID PROTECTION**
  - Insulation
  - Meteoroid Protection
- **DOCKING**
  - Active Ring
  - Passive Ring
- **MAIN PROPULSION**
  - NERVA Engine
  - External Disc Shield For NERVA
  - Purge System & Leak Detection
  - Propellant Management
    - .Propellant Feed
    - .Ground Fill and Drain/Orbital Refueling
    - .Pressurization
    - .Vent
- **AUXILIARY PROPULSION**
  - Reaction Control
- **ASTRIONICS**
  - Inertial Measuring Unit
  - Inertial State Vector Updating Sensors
    - .Position Updating
    - .Attitude Updating
    - .Relative Position
  - Information Management System
    - .Digital Computer
    - .Communications & Tracking
  - Electrical Power
- **SAFETY/ORDNANCE**
  - Safety
  - Ordnance

STRUCTURE

The major elements combining to form the structure subsystem consist of propellant tank, engine thrust structure, skirts, tunnel and fairing, exterior finish and sealer, equipment support structure, astrionics module structure, and aero nose cone structure (expendable).

Propellant Tank

The propellant tank consists of the forward and aft bulkheads and the cylindrical sidewalls. The material employed in its fabrication is aluminum alloy 2014-T6.

Forward Bulkhead

The forward bulkhead is a modified elliptical referred to as oblate spheroid with an aspect ratio of 1.5 and is preferred to a straight elliptical because of its higher volumetric efficiency.

Its geometry in Cartesian Co-ordinates is given by:

$$x = \frac{2a \sin \theta}{(\sin \theta + 1)}$$

$$y = \frac{2a}{3} \left[\frac{\cos \theta}{(\sin \theta + 1)}^{2} - \frac{2 \cos \theta}{(\sin \theta + 1)} \right]$$

where a = one half the major axis

and θ = angle between Y axis and the normal to tangent at the point defined by X and Y coordinates.

Monocoque construction has been chosen for its construction since the design condition is pressure.

Two design conditions were investigated for the bulkhead.

Since the tank is inverted during boost, the bulkhead, in addition to ullage pressure, has to sustain the dynamic head of the propellant which reaches a maximum at S-IC center engine cutoff (CECO). The other condition is the maximum ullage pressure of 27.5 psig at -250°F, the pressurant temperature. Although the temperature of the forward bulkhead is at -423°F during the boost phase, the increased strength at that temperature is not sufficient to offset the difference in pressure between the two cases and therefore the CECO

Figure 5-2. Inboard Profile - RNS Baseline Configuration

5-5, 5-6

SD 71-466-5

condition sized the membrane. Table 5-2 summarizes the loading intensities and the corresponding membrane thickness.

Cylinder Section

Waffle construction (Section C-C of Figure 5-2) with 0-90 degree rib orientation was selected for the stiffening of the cylindrical sidewalls sized for an ultimate loading intensity of 365 lb/in. occurring at Max ($q\ \alpha$) and at LH_2 temperature of -423^oF. The corresponding pressure differential at that condition is 25.0 psig. The design pressure of the membrane is 27.5 psig at -250^oF occurring during space operations. The resulting structural sizes are 0.104 inches for the tank wall thickness and 0.13 inches for the waffle ribs. Circumferential and longitudinal rib pitch are 18.0 and 36.0 inches, respectively, and their height is 1.5 inches. Details of the cylindrical section stiffening are also given in Figure 5.3. Skirt bolting rings have been included at each end of the cylindrical section to provide for the attachment of the skirts as shown in details B and D of Figure 5.2.

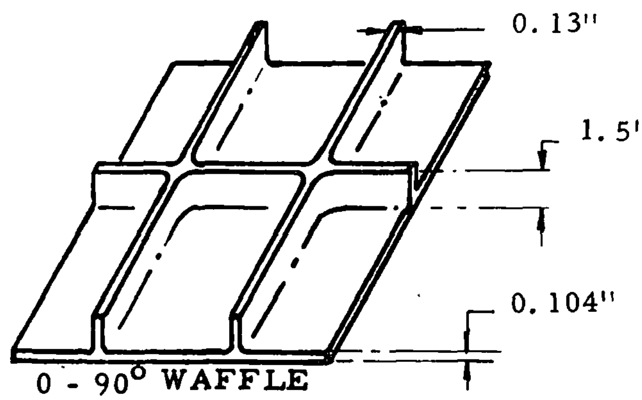


Figure 5-3. Tank Sidewall Construction

Aft Bulkhead

As for the forward bulkhead, the construction of the aft bulkhead is essentially monocoque since the primary design condition is pressure (bulkhead depleted, pressure = 27.5 psi and temperature = -250^oF). Table 5-3 summarizes the loading intensities and the corresponding membrane thickness for the latter condition. A bolting ring is incorporated at the intersection with the thrust structure to attach the latter to it. Detail of this assembly is given in the main view of Figure 5-2.

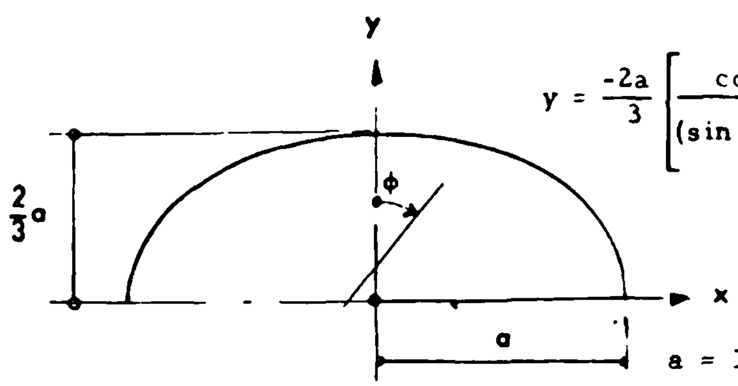
Table 5-2 Loading Intensities and Wall Thickness for the Oblate Spheroid Forward Bulkhead

Condition	Max. Acceleration @ -423°F			Max. Ullage Pressure @ -250°F		
Station ϕ^o	Meridional Load N_ϕ (lb/in)	Hoop Load N_θ (lb/in)	t_w (in)	Meridional Load N_ϕ (lb/in)	Hoop Load N_θ (lb/in)	t_w (in)
0	9483	9483	0.105	7587	7587	0.104
20	7083	4623	0.078	5680	3737	0.078
40	5763	1972	0.064	4640	1657	0.063
60	5055	528	0.056	4085	547	0.056
80	4741	-165	0.053	3840	58	0.053
90	4703	-264	0.161*	3811	0	0.161*

* Thickness based on weld land. The material thickness tapers between stations.

$$x = \frac{2a \sin \phi}{(\sin \phi + 1)}$$

$$y = \frac{-2a}{3}\left[\frac{\cos \phi}{(\sin \phi + 1)^2} - \frac{2 \cos \phi}{(\sin \phi + 1)}\right]$$

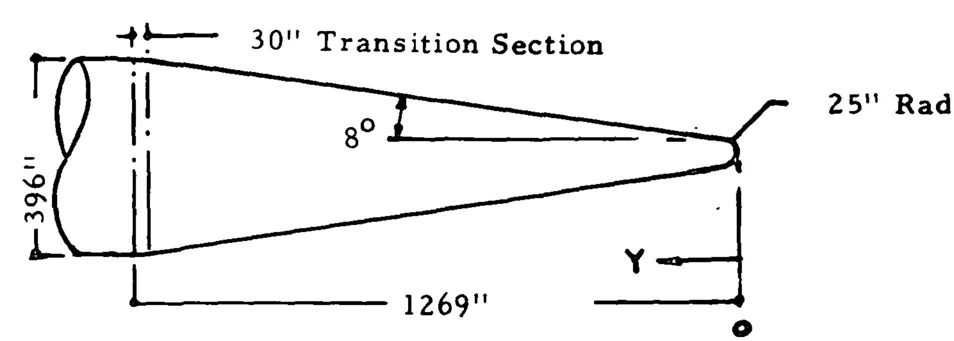


a = 198.0"

Table 5-3i. Loading Intensities and Wall Thicknesses for the Conical Aft Bulkhead

Condition	Max. Ullage Pressure @ -250 F		
Station Y (in)	Meridional Load N_ϕ (lb/in)	Hoop Load N_θ (lb/in)	t_w (in)
0	481.25	481.25	0.030
21	481.25	481.25	0.030
110	740.00	1454.	0.030
260	1176	2274.	0.031
410	1624	3093	0.042
560	2088	3913	0.053
710	2568	4733	0.064
860	3067	5552	0.076
1010	3589	6372	0.087
1160	4136	7191	0.098
1269	4479	7642	0.161*

* Thickness based on weld land. The material tapers between stations.



Engine Thrust Structure

The thrust structure shown in Figure 5-2 is a cone frustum approximately 107 inches long to which is attached the active docking ring of the neuter dock concept developed by the NR/SD Space Station study for potential application to interfacing space program elements. Because of thermal considerations the material selected for the thrust structure is 6Al-4V titanium alloy which has a substantially lower conductivity than aluminum. Boron epoxy and fiber glass would be superior in this application; however, the vicinity of the thrust structure to the NERVA and to the consequent high-radiation dose rates precludes their use until structural degradation of these materials is assessed. The upper edge of the thrust structure is bolted to a ring integral with the aft conical bulkhead. The construction selected is ring-stiffened with the basic skin, t_{sk} = 0.10 inches.

Forward Skirt

To minimize heat leak to the tank the forward skirt incorporates a 4-foot heat block consisting of 0.04 inches fiber glass faced sandwich with a 2.0 inch HRP honeycomb core. The heat block is located next to the tank attach point as shown in detail D of Figure 5-2.

The basic construction selected for the skirt structure is integrally, longitudinally stiffened aluminum alloy 7075-T73, with 1.50 inches uprights at a 4.32-inch pitch. The frames required with this construction are mechanically fastened to the inside face of the skin at a pitch of approximately 30 inches. The stiffener pitch of 4.32 inches has been chosen for compatibility with the bolt spacing and stiffener distribution of the S-II forward skirt and interface joint. This is necessary to maintain load path continuity during boost. The upright thicknesses vary from 0.165 to 0.20 inches while the skin thickness tapers from 0.11 to 0.133 inches. The frame cross section is approximately 2.0 square inches.

Aft Skirt

The same basic construction is used for the aft as for the forward skirt. A heat block with the same dimensions and for the same purpose i.e., heat leakage mitigation to the tank, is employed. The heat block is located next to the tank attach point as shown in detail B of Figure 5-2. To the aft edge of the heat block is attached a frame which doubles as aero nose cone support and as an anchor point for stage transportation and KSC operations.

Tunnel and Fairing

A systems tunnel running the length of the stage is shown in Figure 5-2. Fairings at the forward and aft skirts are used to smooth out the air flow over the tunnel on the cylindrical portion of the stage. The tunnel houses the autogenous pressurization line and the sundry electrical wires and coaxials cables to the NERVA, thermo-dynamic vent, and ordnance for the propellant dispersion.

As presently projected, based on S-II experience, the fairings consist of fiberglass sheeted half cones supported by formers attached to the skirts. The tunnel is hemispherical in cross-section and hinged about one of its edges to permit access for inspection and repair of the components within it.

Exterior Finish and Sealer

All exposed surfaces are painted with a white epoxy polymide air drying primer while the closed-cell foam applied to the tank wall for cryo pumping prevention, receives a sealing coat of chem seal polyurethane and a finish coat of dynatherm. These finishes are a direct result of analysis and tests on the foam employed over the S-II LH_2 tank in typical Saturn V ground hold and boost environments.

Equipment Support Structure

Under this designation is included the aluminum alloy built-up construction employed mainly in the forward skirt and Astrionics bay area to support the GN&C, electrical subsystem, RCS bottles cradles and the docking cone, plus the tracks to facilitate removal and replacement of components and sundry clips and brackets. The majority of these items have been identified in Figure 5-2, albeit in a sketchy manner. Further definition of this area will be accomplished as the loading environment and finalization of equipment size and location are established.

Astrionic Module Structure

The Astrionics Module structure is 60 inches high and can be considered integral with the forward skirt since it is an extension of it. The construction is therefore integrally, longitudinally stiffened aluminum alloy 7075-T73, with 1.5 inches uprights at a 4.32 inch pitch. As discussed in the forward skirt section the latter is based on compatibility with the bolt spacing and stiffener distribution of the S-II forward skirt and interface joint. Frames are located internally at approximately 30 inch pitch.

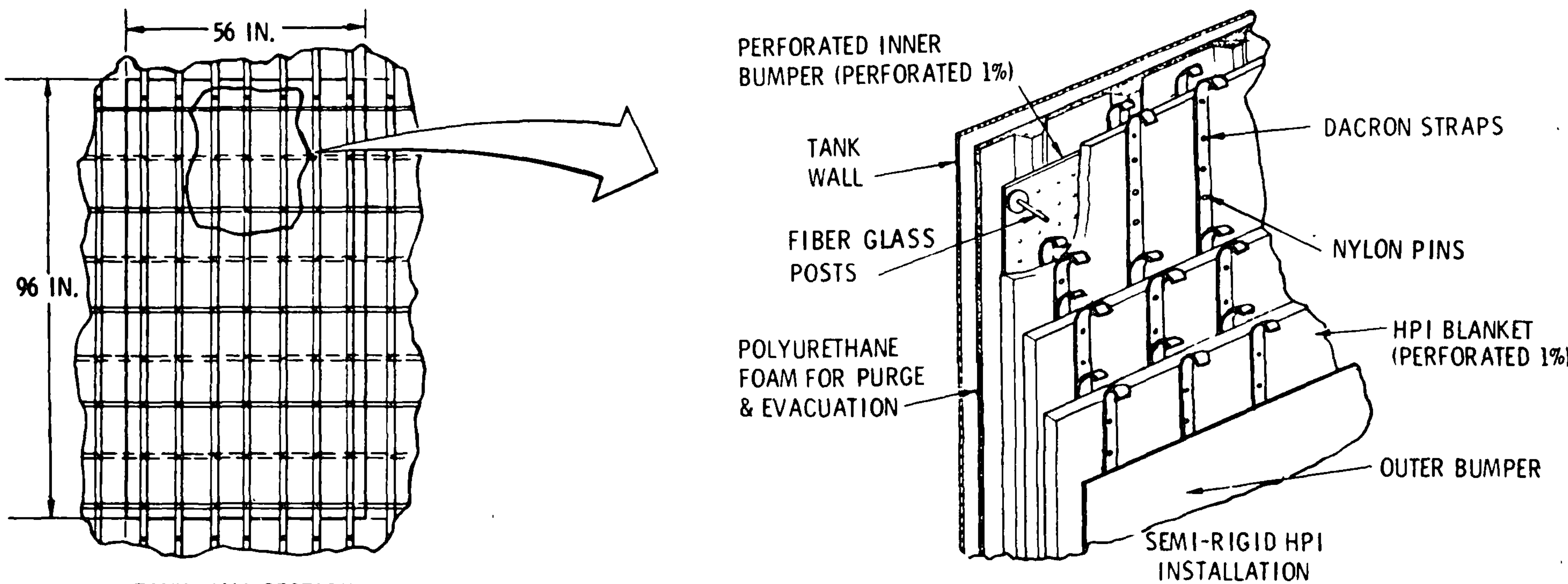
Aero Nose Cone

The nose cone cover shown dotted in Figure 5-2 is employed during the boost phase through the sensible atmosphere to protect the meteoroid/ thermal protection over the conical aft bulkhead from thermal as well as dynamic pressure loading. It may be jettisoned once the q loading drops below 1.0 psf; however, reuse considerations would require that the shroud be carried to orbit, dismantled and returned via Space Shuttle. The construction employed for its fabrication is honeycomb sandwich with 0.079-inch fiber glass facings and a 3.0-inch thick core. It is attached to the aft skirt edge member by either explosive bolts or quick release fasteners, depending on whether it is expendable or recoverable. One mode of disposal is to cut the cone into gores which would in turn deploy and be forcefully ejected away from the vehicle.

THERMAL/METEOROID PROTECTION

The thermal/meteoroid protection subsystem covers most of the exposed surfaces of the vehicle, as shown in Figure 5-2 and detailed in Section C-C, and details B and D. Definition of the installation is also given in Figures 5-4 and 5-5 which represent integrated HPI/meteoroid protection system design wherein HPI functions not only to thermally protect the hydrogen but to form a plurality of meteoroid shields to increase the efficiency of the meteoroid protection system. In addition, the integral nature of these concepts provide (1) protection for the HPI from aerodynamic heating and wind loads, (2) dynamic damping of the meteoroid bumpers, (3) a sealed area for the distribution and release of HPI purge gases during the ground hold and launch phases of the mission, (4) a more stable region for structural support of the HPI than afforded by the tank wall which undergoes dimensional changes due to internal pressures and cryothermal contraction, and (5) ease of manufacturing, installation, inspection, and repair of the system.

The double bumper concept, employed over the tank sidewalls, consists of two layers of fiber glass 0.030 and 0.010 inches thick for the outer and inner shields, respectively, encasing 1.5 inches net of GAC-9 in three separate panels, 0.5 inches each. The design is as shown in Figure 5-4. Both bumpers are continuous cylinders with the outer sheet beaded to impart stiffness to prevent local flutter. The two sheets are kept apart by rows of radial fiber glass posts spaced approximately 8.6 inches circumferentially and 8 feet longitudinally. The insulation panels "hang" from these posts. To avoid imposing excessive bearing loads on the "flimsy" aluminized mylar layers of the HPI, dacron straps are used on either side of the panel at each post location. These straps are used to discretely pick up the weight of the insulation at the density



TANK WALL SECTION
(HPI MODULE)

Figure 5-4. Double Bumper Concept

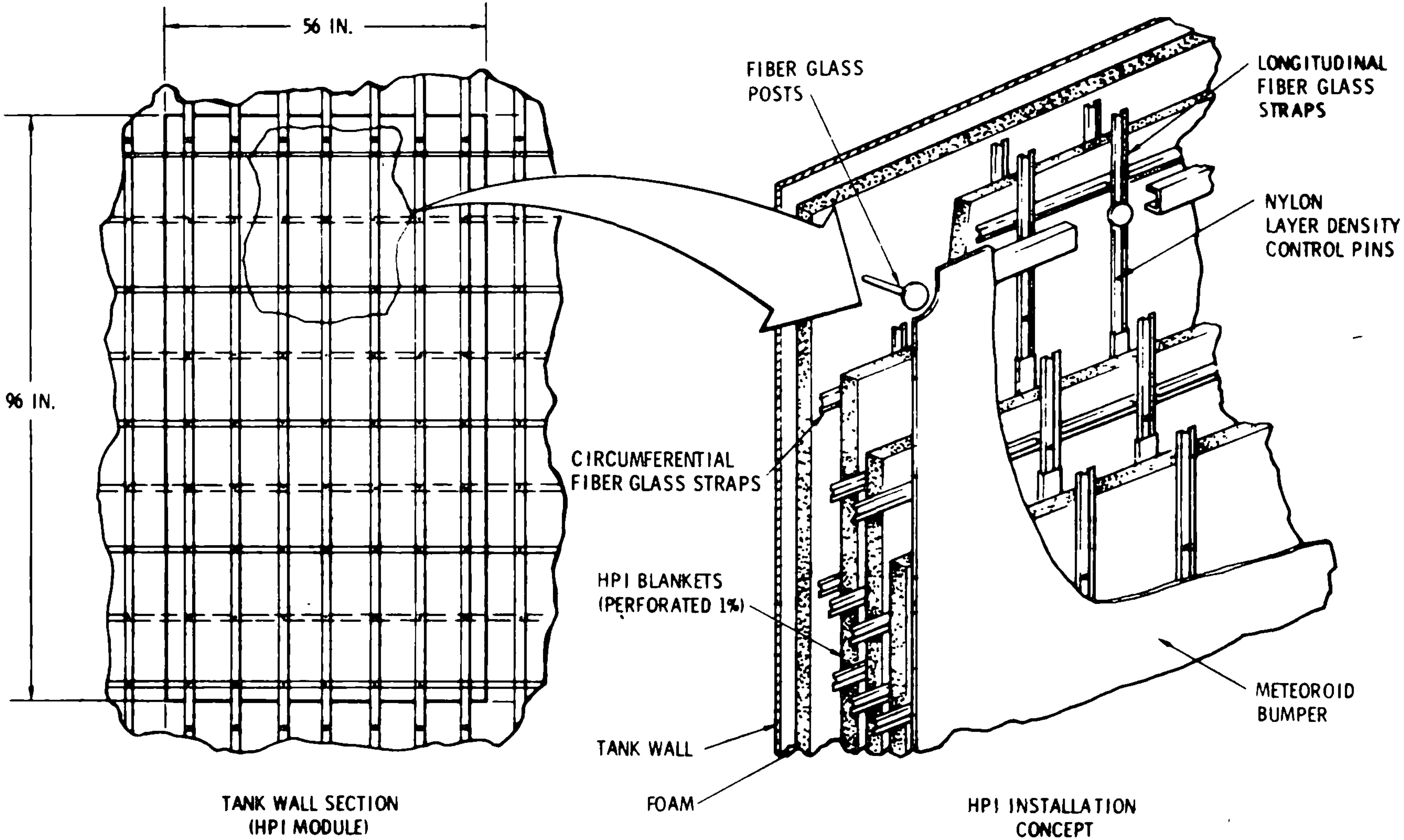


Figure 5-5. Single Bumper Concept

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control pins. The latter consist of 30 mil nylon pins which are spaced longitudinally in line with each post and to which is applied a thin adhesive coating. By special techniques developed at NR/SD precision perforations in the HPI panels are formed without incurring piercing or tearing of the insulation. The pins are then inserted in the perforations and bonded to the individual HPI layers of the panel in a single operation. The pin ends are bonded to the dacron straps.

The inner bumper is perforated to allow the purge gasses to escape into vertical channels precut in the tank sidewall closed cell sprayed foam.

The thermal/meteoroid protection over the conical aft bulkhead is a single bumper design as shown in Figure 5-5. Purge gasses in this instance are evacuated through the annular space formed between panels. This is accomplished by substituting the dacron straps with rigid fiber glass epoxy molded straps.

Figure 5-2 shows the external thermal meteoroid protection system subdivided into three segments, i.e., a short cylindrical segment over the engine thrust structure attached to the aft end of the latter, a conical segment over the aft bulkhead attached to the aft skirt, and a cylindrical segment covering the balance of the external surface including the skirt heat blocks and attached to the aft end of the aft skirt. Additionally, insulation meteoroid protection blankets are provided over the forward bulkhead around the inside of the forward and aft skirt, and over the aft bulkhead cap and around the inside of the engine thrust structure with attachment to the skirts only. This arrangement eliminates direct heat paths between the insulation and the tank wall, yet accommodates the relative thermal deflections between tank and shell.

DOCKING

The stage interfaces with the NERVA at the aft end and with other potential space program elements at the forward end. These elements include Propellant Depot, Maintenance Element, Space Tug, and payload. The Neuter Docking system developed by NR/SD under the Space Station Study is employed at both ends of the stage to maximize commonality.

The docking system consists of an active assembly and a passive ring. Active assemblies are mounted to the docking cone at the forward end of the stage, and to the thrust structure at the aft end. A passive ring is attached to the NERVA forward thrust structure to mate with the stage in orbit and to facilitate engine removal and disposal, if required.

Active Ring and Cone Assembly

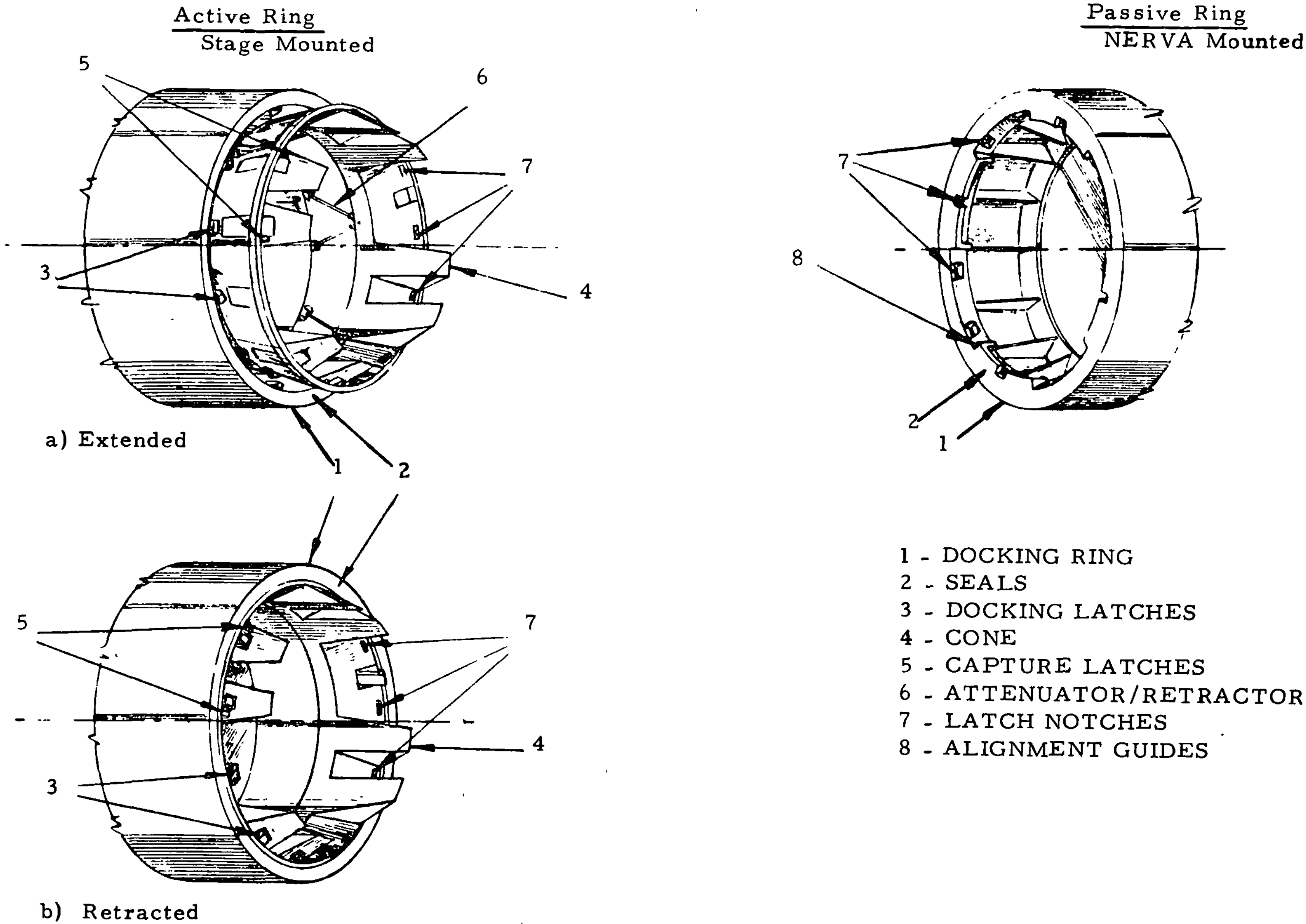
This assembly is shown in Figure 5-6 and consists of a ring, cone, capture latches, attenuator and retractor cylinders, docking latches, and docking seals. The cone is attached to six pairs of pneumatically operated attenuator and retractor cylinders that provide a 10-inch stroke. The cone is slotted to provide fingers which are spaced and tapered so that an identical, approaching cone will mesh with it. The intermeshing tapered fingers provide radial and angular indexing capability as they mesh. The capture latch subassemblies consist of two pair of diametrically opposed latches that are mounted in recesses in the cone. Four keyed notches also are located in the cone so that the cone will capture an approaching cone or passive ring. The capture latches serve to tie two cones or a cone and a passive ring together until retraction occurs and the docking latches are engaged.

A docking ring which is built (welded or bolted) into the vehicle structure provides the docking seal surface. Dual seals fit into grooves machined into this surface. The docking ring also provides the structural attachment points for the attenuator and retractors and the docking latches. The notches into which the docking latches of a mating ring fasten also are provided inside the docking ring.

When the attenuator and retractors are actuated, the cone is retracted into the docking ring. The docking ring inside surface is tapered and a matching taper exists on the cone. These surfaces come together to provide final alignment and shear capability. The 12 independently operating automatic docking latches are then engaged to lock the docking rings together.

Passive Ring Assembly

The passive ring assembly, also shown in Figure 5-6, does not have any active components. It is a machined aluminum alloy ring having a docked seal face and a tapered inside surface with notches, to receive the cone tapered fingers, the capture latches, and the docking latches of the mating active ring and cone assembly. The docking ring is welded or bolted to the adjoining and supporting structure.



1 - DOCKING RING
2 - SEALS
3 - DOCKING LATCHES
4 - CONE
5 - CAPTURE LATCHES
6 - ATTENUATOR/RETRACTOR
7 - LATCH NOTCHES
8 - ALIGNMENT GUIDES

Figure 5-6. Neuter Docking System

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MAIN PROPULSION

This section includes the NERVA engine and its external radiation disc shield for manned missions, and stage related subsystems required for the proper operation of the vehicle. The latter are covered under Propellant Management and include the propellant feed, pressurization, fill and drain, and venting subsystems.

NERVA Engine

The NERVA engine, illustrated in Figure 5-7 , utilizes a nuclear reaction to provide heat to liquid-hydrogen propellant in a "full-flow" cycle and thus obtain a high-specific-impulse propulsive force. In the full-flow cycle, all propellant (including turbine drive fluid) passes through the reactor core and the nozzle to produce thrust, thus maximizing specific impulse.

Thrust is generated by hydrogen which is heated under pressure in the reactor core and expelled through a De Laval nozzle. The hydrogen is supplied from the tank by means of dual turbopumps and flows through cooling passages in the nozzle wall and reflector before reaching the reactor. Energy transferred to the hydrogen in the cooling passages is extracted by the full-flow turbines to drive the pumps. Bypass valves control the output of the turbopumps and thus regulate chamber pressure. Reactor power is regulated to maintain a desired chamber temperature by control drums and by structural support coolant valves (which control the hydrogen density in the reactor support structure).

The primary design requirements include the capability of providing nominal rated thrust and specific impulse of 75,000 pounds and 825 seconds, respectively, and a 10-hour operating life at rated temperature, accumulated in as many as 60 operating cycles. The engine is required to perform with high reliability and safety for man-rated applications.

The two basic elements of engine control are turbine power and reactivity. Both are used in the control of the primary engine variables: chamber temperature and pressure as demanded by the engine programmer. Input to the programmer comes from the stage or ground-test control and is typically the start command, set points, shutdown, and emergency actions. The engine reliability allocation is 0.995 and is to be man-rated requiring that maximum effort in the design be directed towards the elimination of single failures or combinations of failures that could endanger personnel, including the launch crew and general public.

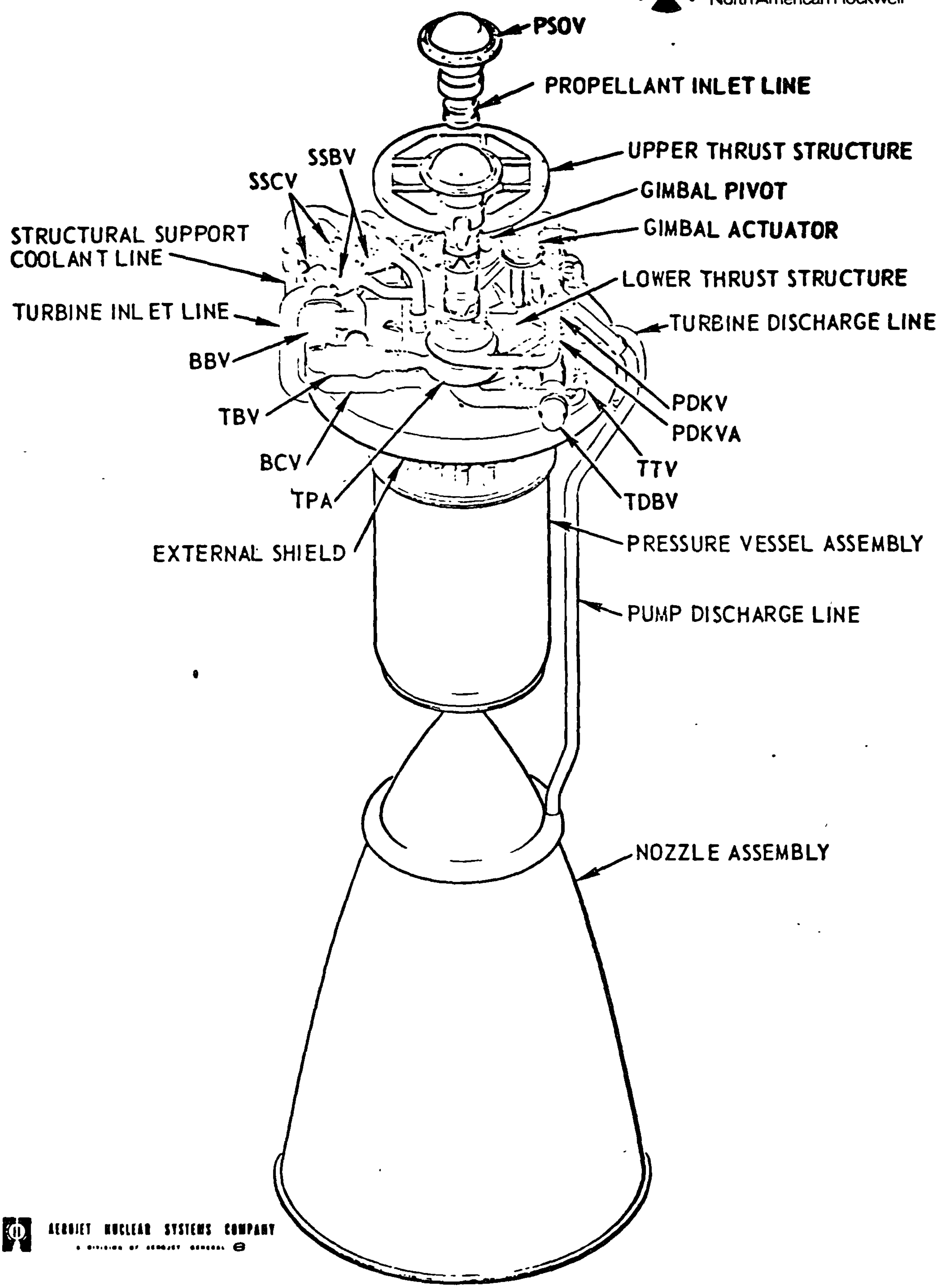

Figure 5-7 1137400C Engine Concept

Diagnostic instrumentation, adequate to detect deteriorating situations or incipient failures, will be selected and the ability to override the engine programmer remotely by the crew and ground control (as well as the capability for remote thrust reduction independent of the engine program) will be provided. Additionally, the engine will incorporate means of preventing accidental criticality during all ground and space operations. An anticriticality destruct system will be provided for launch and ascent. Also, the engine is being designed for maintainability both on the ground and in space. Radiation shielding for protection of crew and passengers is designed for removal and reinstallation in space. The reference design provides a single separation plane to permit the engine replacement on the stage. In addition, the various subsystems of the engine are being modularized to simplify replacement of failed components.

Safety requirements of the overall flight system dictate the need for a second turbopump leg or branch to provide redundancy of the active functional elements of the engine. Provision also is made for isolating each leg from the system in case of a malfunction. For reliability, additional bypass control valves are added so that in the case of a failure of a single valve, the redundant turbopump leg will not fail also. Redundant valves are also provided in the cooldown line and the structural support coolant system.

Not shown in the figure is the neuter passive docking ring, described under Docking Subsystem, which is attached to the upper thrust structure of the NERVA.

The current calculated and projected weights of the engine are presented in Table 5-4. For unmanned mission applications, the engine will not be equipped with an external shield. The weights stated apply to the engine in the operating state (without the destruct subsystem). They do include a projected weight allocation of 500 pounds for the stage-mounted portion of the NERVA digital instrumentation and control electronics, located forward of the primary engine-stage interface. The weight employed in the stage definition is the current calculation shown in the table.

Table 5-4. Nerva Engine Weight Summary

	Current Calculation	Target
Engine weight, excluding external shield	27, 728 lb.	23, 500 lb.

Estimated locations of the engine center of gravity (C.G.), with and without external shield, are shown in Figure 5-8. The C.G. locations are for the gimbaled engine, exclusive of all equipment forward of the gimbal point.

Preliminary values of moments of inertia of the dry engine, in operating configuration, taken about the gimbal point (X = 23.0 in., Y = 0 in., Z = 0 in.) and using calculated weights, are presented in Table 5-5.

Table 5-5. Moment of Inertia About Gimbal Point

	With External Shield	Without External Shield
Roll axis, slug-ft^2	6,275	3,800
Pitch axis, slug-ft^2	90,723	83,618
Yaw axis, slug-ft^2	90,692	83,587

The gimbal assembly subsystem provides a capability for adjusting the thrust vector angle by at least +3 degrees in any direction. Of this, ±1.5 degrees is allocated to vehicle misalignment compensation. The rate capability of the subsystem is 0.25 deg/sec and the acceleration capability is 0.50 deg/sec^2. The actuators are electrically operated and respond to position command signals received from NERVA digital I&C electronics system.

The engine is designed for a minimum useful life in various categories, as shown in Table 5-6, and is required to perform without degradation of reliability, performance, and endurance subsequent to the storage designated.

Table 5-6. Nerva Useful Life

Life Category	Requirement
Operating Life at 4250°R T_c	500 min (minimum) (60 cycles)
Space Life (Operating and Nonoperating)	3 years
Ground Storage and Pre-Launch Operations	
Storage (controlled environment)	5 years
Launch Pad Environment	6 months

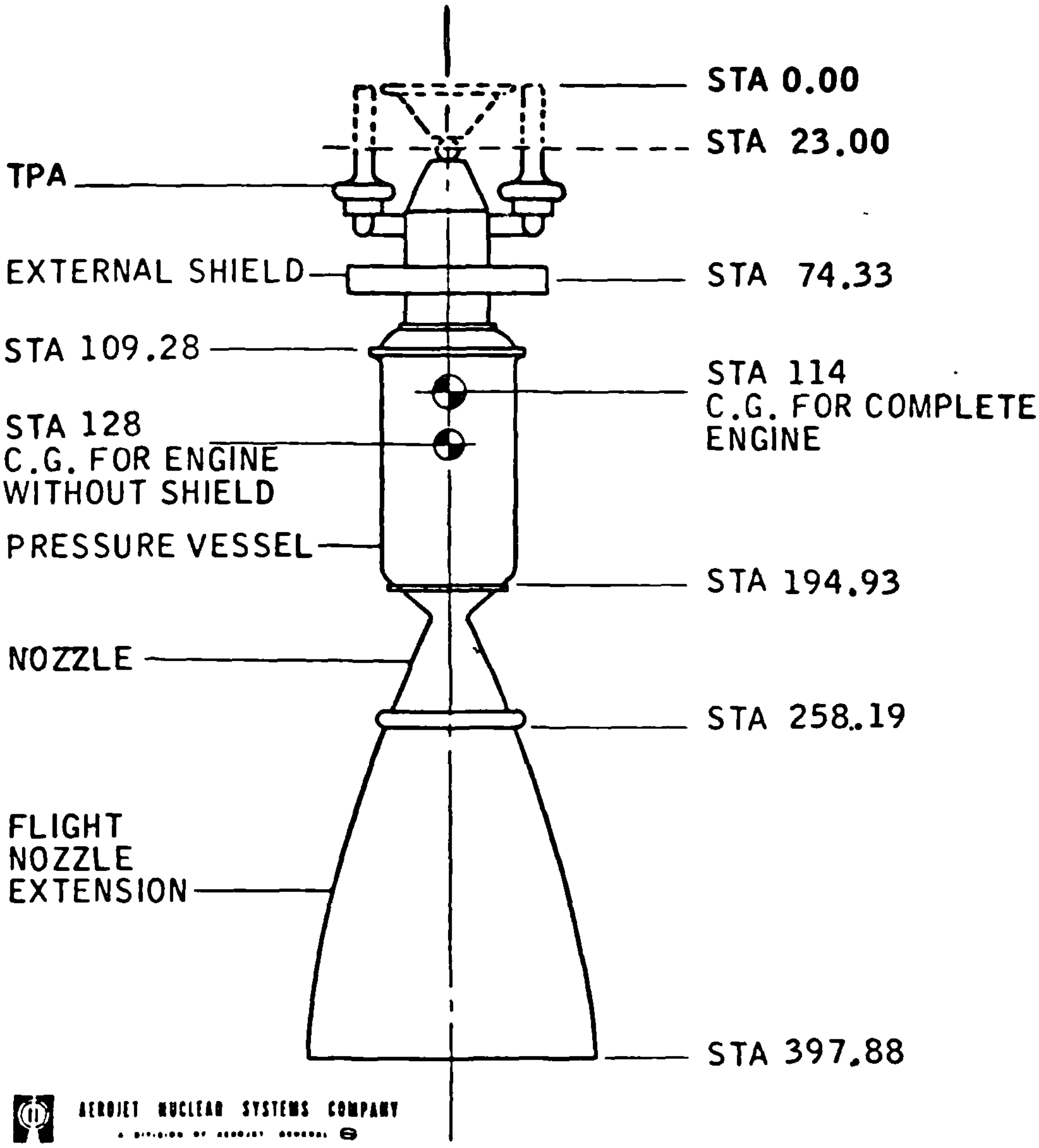

Figure 5-8 Approximate Center of Gravity Locations for Gimbaled Flight Engine

External Disc Shield for NERVA

The weight for the external disc shield to satisfy the 10 REM integral tank top dose criterion, is 4050 pounds with a 44-inch disc radius. This result is based on G 4 SCAT and WAND-REV computations based nominally on the CRAM model 10,000 pound, 50-inch radius external disc shield design.

Purge and Leak Detection

During the prelaunch, hold and boost to earth orbit, the HPI has to be maintained in an inert, dry atmosphere. One major reason for this conditioning is that moisture will tend to remove the aluminized layer over the plastic film, whether mylar or kapton, and any other contaminant on the surface of the HPI may cause continuous outgassing with a consequent degradation of the insulation properties. Therefore, circumferential inlet manifolds are located at the edge of each major segment of the Meteoroid/Thermal protection subsystem to diffuse nitrogen gas within the insulation. These manifolds are connected to ground support equipment lines for the pumping of GN_2 at pressures above ambient to maintain an outflow of gas at the exit. At the other end of each major segment of the Meteoroid/Thermal protection subsystem a gas permeable membrane is used to allow an even distribution throughout the HPI of the inert gas. The pressure differential and gas flow have not as yet been identified. Further work is needed in this area to assist in sizing of the manifolds.

Propellant Management System

The approach taken in establishing operational techniques and system design for management of the stage propellant has been to treat hydrodynamics and thermodynamics as a single, integrated study task. This approach is dictated by the mission requirements and the physical and thermodynamic properties of liquid hydrogen.

A passive propellant management which exploits the inherent characteristics of the hydrogen, and the shape and size of the RNS is recommended. The characteristics utilized are the large stage volume, high fineness ratio, large ullage created by the TLI and LOI burns (2/3 of propellant utilized), ullage stratification during pressurization (for a diffuser-type pressurization inlet), and the natural tendency for hydrogen to stratify because of heat leak. Passive propellant management utilizes vertical ullage temperature gradients that occur during pressurization and the slow approach to thermodynamic equilibrium (long conduction path, small gradients) to reduce boiloff and burnout residuals. Combined with a judicious propellant feed system, the integrated design results in improved thermal control and overall operational simplicity. The capillary devices, discussed below under Propellant Feed System and used for propellant acquisition and location control are also used to mitiggate active mixing of ullage vapors and liquid hydrogen. In the same vein, such devices as liquid mixers and ullage sprays, which attempt to maintain thermodynamic equilibrium, have been purposely avoided.

The resultant fluid system presented schematically in Figure 5-9 provides an overview of the propellant feed, pressurization, fill and drain, and vent subsystems. Only those details of the NERVA fluid system which have a direct interrelationship with the stage propellant feed, cooldown, and pressurization subsystems are shown. Also included are details of the ground and orbital fill subsystems. Table 5-7 identifies the numbered components in the schematic.

Some components of the fluid system are utilized for dual functions. For example, the orbital fill line (5) is used as a vent line during ground fill and the emergency vent valves (17) are used as fill valves during the ground fill operation. During orbital fill, the LH_2 fluid flow is controlled by the on-off orbital fill valve (2) and the orbital fill proportioning valve (3) distributes the flow between the spray head (4) and orbital fill line (5). The sprayed LH_2 collapses the ullage and liquid LH_2 is delivered to the bottom of the RNS tank downstream of the cooldown compartment capillary barrier.

The thermodynamic vent subsystem is comprised of in-orbit vent valves (9), vent throttling valves (10), propellant conditioning flow proportioning valves (11), heat exchanger tubes (12), vent flow rate regulators (15), and non-propulsive vents (16). LH_2 is drawn through the cooldown line, just upstream of the cooldown flow valves (8), expanded through the throttling valves, passed through the heat exchangers to pick up heat from the propellant and ullage gas and finally expelled through the vent flow regulators and non propulsive vents.

Propellant is fed to the NERVA engine through two propellant feed lines and controlled by the propellant shutoff valves (7). Cooldown propellant is supplied through the cooldown line and is controlled by the cooldown flow valves (8).

RNS tank pressurization gas is provided by the NERVA engine during engine run through the tank pressurizing line (19). Tank pressurant flow is controlled by on-off flow valves (22) and small (20) and large (21) orifices.

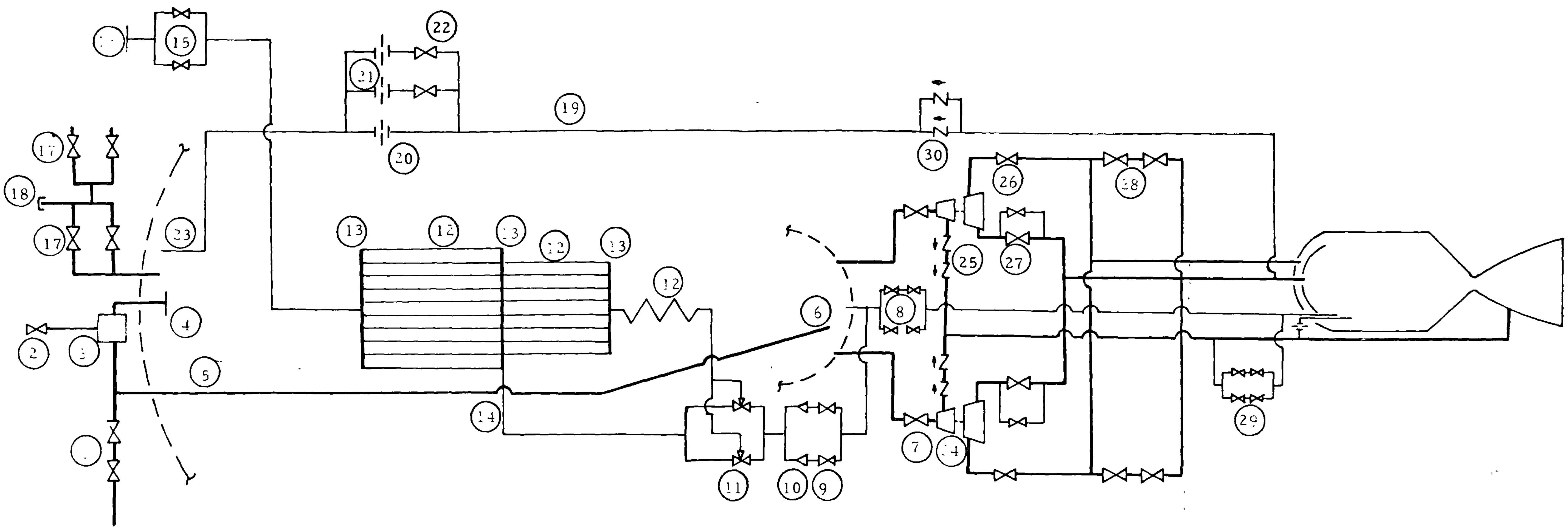

Figure 5-9 RNS Fluid System Schematic

Table 5-7. Fluid System Components

1. Vent Valve for Ground Fill
2. Orbital Fill Valve On-Off
3. Orbital Fill Proportioning Valve
4. Spray Head
5. Orbital Fill Line
6. Orbital Fill Line Diffuser
7. Propellant Shutoff Valve
8. Cooldown Flow Valve
9. In-Orbit Vent Valve, On-Off
10. In-Orbit Vent Throttling Valve
11. Propellant-Conditioning-Flow Proportioning Valve
12. Heat Exchanger Tubes (outlet region, conical section, and cylindrical section)
13. Heat Exchanger Manifold
14. Heat Exchanger By-Pass
15. Vent Flow Rate Regulators
16. Non-Propulsive Venting
17. Emergency Vent On-Off Valve Quad
18. Ground Fill Line (capped off)
19. Tank Pressurization Line
20. Small Orifice
21. Large Orifice
22. On-Off Flow Valves
23. Pressurant Inlet Diffuser and Baffle
24. Turbopump Assembly
25. Check Valves
26. Turbine Block Valve
27. Turbine Discharge Block Valve and Turbine Throttling Valve
28. Bypass Block Valve and Bypass Control Valve
29. Structural Support Block Valve and Control Valve
30. Stage Pressurant Check Valve

Propellant Feed

The propellant feed supplies propellant at the required flow rates, temperatures and pressures during the various phases of engine operations. The hardware associated with the system consists of the capillary devices and feed lines described below.

Capillary Devices. These have been developed to provide: (1) feedout during restart, steady burn, and cooldown; (2) slosh control; and (3) thermal control, venting and pressurization. They divide the tank into four major compartments and are named to denote their major function, i.e., (1) ullage compartment, (2) bulk propellant compartment, (3) cooldown compartment, and (4) restart compartment. These are shown in Figure 5-2 with details in Figure 5-10.

The bulk propellant capillary barrier is mounted and supported at the intersection of the cylindrical and aft bulkhead at a plane just above the propellant level after the TLI burn. The barrier consists of two perforated plates half inch apart with 0.060 inch chemically milled holes. The fraction of open area to total plate area for each plate is 0.1. The plates are supported by radial and circumferential frames, conically shaped for structural purposes, with the apex located two feet above the base. This barrier will prevent all but inconsequential gas-liquid interchange between compartments for rotation rates and lateral, negative and centrifugal accelerations due to vehicle maneuvers and perturbations. The perforations diameter and cone height are based on the solution of a stability equation for the retention of propellant against lateral and rotational maneuvers with the former being the controlling parameter. The gap between the plates provides a wicking and resealing capability. A 0.5-inch separation is recommended based on information available at the present time.

The restart and cooldown compartments assure adequate propellant flow at each engine restart. Hole size and fraction openness differ from that of the bulk capillary barrier. They are respectively 0.020 inches and 0.45 openness fraction for the cooldown compartment, 0.025 inches and 0.35 openness fraction for the restart compartment. The cone height of the cooldown compartment capillary barrier is 3 feet. Each compartment is thermally cooled by a thermal conditioning unit which is part of the "thermodynamic" vent system. The restart compartment, which has a capacity of 500 pounds, supplies vapor-free propellant at each engine restart. This compartment is also designed to assure adequate feedout for the last phase of the last engine cooldown. During cooldown propellant is fed from the restart compartment to the collector compartment (Section A-A of Figure 5-10).

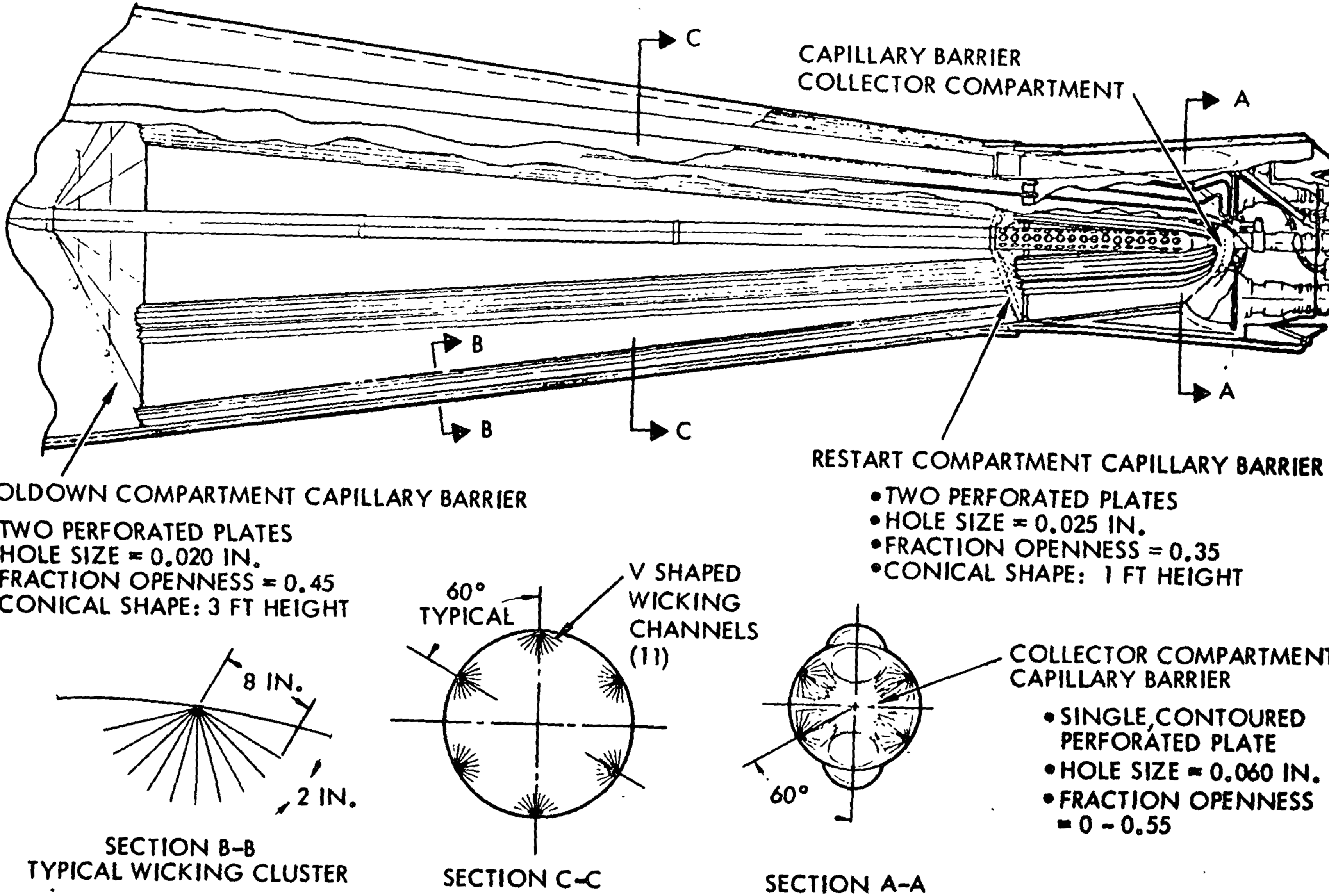

Figure 5-10. Propellant Acquisition & Location Control Configuration

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The cooldown compartment supplies the bulk of the cooldown propellant during the mission and is sized for "worst case" hydrodynamics for the lunar mission. As the bulk compartment is partially depleted, propellant could be dislocated to the upper end of the compartment which would lead to vapor passage from the bulk to the cooldown compartment as cooldown flow proceeds. To preclude vapor passage to the restart compartment, six V-shaped wicking clusters have been added and are shown in the main view and in Section C-C. These clusters are sized to provide cooldown flow to the restart compartment under zero to minus 10^{-5} g's. The size and shape of the wicks, Section B-B, have been determined using data from NR in-house studies (Reference 5.1).

Feedout during the last cooldown requires wicking clusters in the restart compartment also (Section A-A). Therefore, four of the V-shaped clusters are extended to lead fluid into the collector. V-shaped channels are specified as they are self-emptying, thereby, reducing trapped residual in the compartment. The collector capillary barrier is contoured to the aft bulkhead cap geometry with cut-outs to avoid the two pump outlet lines. Fraction openness is designed to increase from 0 to 0.55 with increasing distance from the centerline. This is done to prevent vapor pull-through (interface dip) over the outlet line. Perforation size is 0.060 inches.

Feed Lines. A schematic of the feed flow is shown in Figure 5-11. Feedout is accomplished by two 9.5-inch diameter lines connected through PSOV valves (16) to the two NERVA engine turbopumps. The cooldown propellant is supplied through a 3.0-inch diameter line with the flow controlled by the cooldown flow valve (17).

Ground Fill and Drain/Orbital Refueling

Since the tank is in an inverted position during boost, ground fill and drain are accomplished via line (1) as shown in Figure 5-11. Propellant is pumped in via this line and pumped out (using a different pump), should ground draining be required. A 3.5-inch diameter line is adequate for this purpose. Just prior to launch, this line is capped off. To assure that inadvertant drainage or venting does not occur during the flight, the valves (2) are normally maintained in the closed position. The ground fill and drain system utilizes, in common, some of the lines of the refueling or emergency vent system. This system will be discussed subsequently.

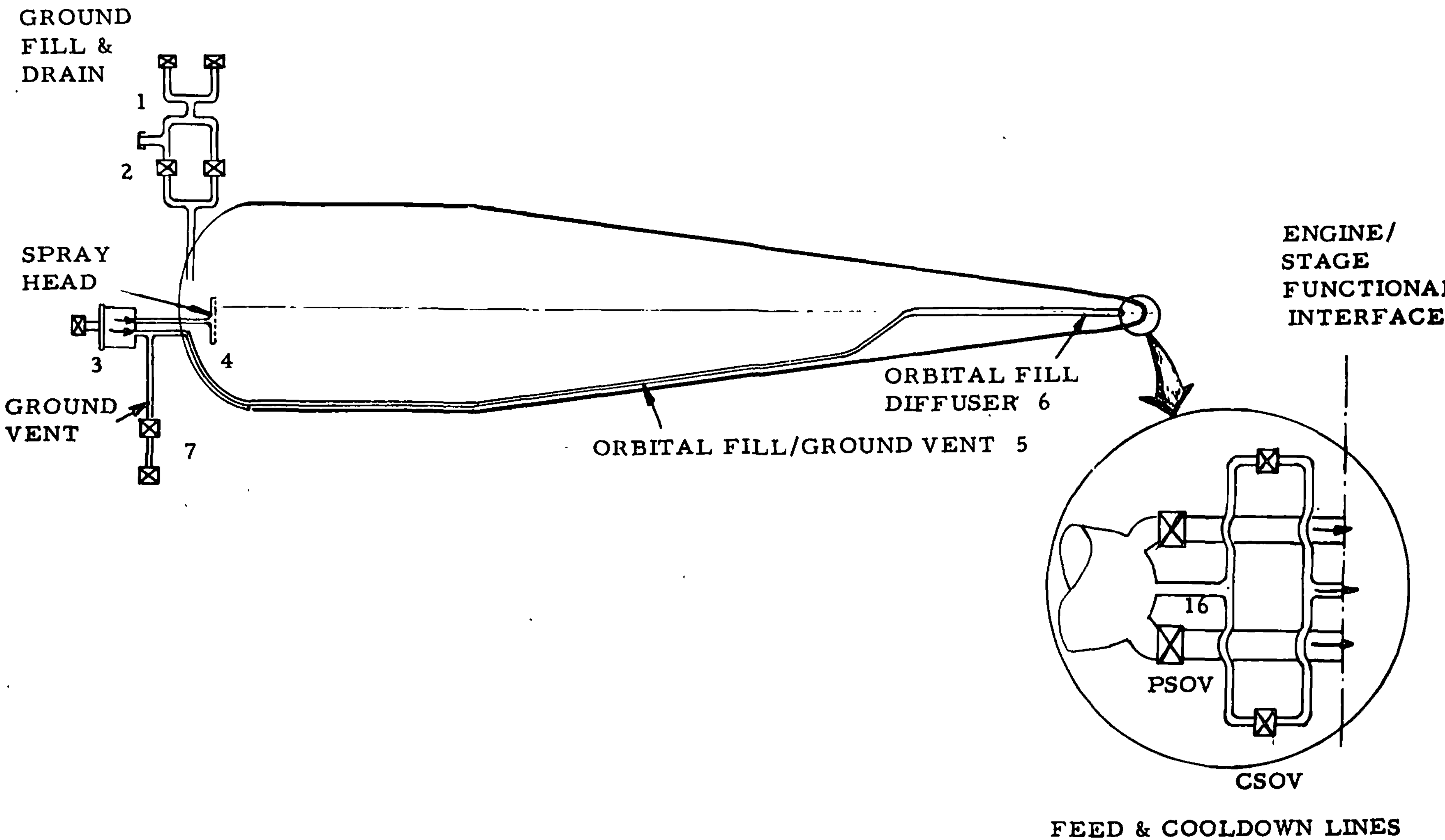

Figure 5-11. Propellant System Schematic

An artificial gravity field for propellant positioning using linear or rotational acceleration was recommended in Volume II A, Section 3 - Orbital Operations, to facilitate refueling in orbit. Refueling in the no vent mode (locked tank) is no longer required and the minimum tank pressure at the first engine start has been lowered to 15.5 psia. Therefore, the system proposed allows filling in the no vent mode or can be used with venting for conditioning of the tank and its contents.

As shown in Figure 5-11, filling in the no vent mode utilizes a divider valve (3) which proportions the entering propellant between the fill line (5) and the ullage spray (4). If tank pressure should rise excessively, flow is diverted to the ullage spray (4) to reduce tank pressure, the divider valve operating off tank pressure. However, in order to assure that the fueling process does not take excessive time and that the final temperature of propellant is sufficiently low ($\sim$36.8°R), it is likely that venting prior to or during refueling will be required. This can be accomplished by opening the refueling/emergency vent valves (2). The artificial gravity field and the capillary barriers will allow liquid-free venting through this system throughout most of the refueling. The refueling/emergency vent system (1) can also be used to return vapor to the supply tank. This refueling approach, which has been recommended in the NR Propellant Depot Study for NASA-MSFC, requires that propellant be pumped from the source tank to the RNS to maintain a higher pressure in the latter.

The orbital refueling system has another important feature, the reduction and dissipation of inlet momentum. As it is important that this momentum be reduced to prevent structural damage and vehicle perturbations, the proposed design utilizes an inlet diffuser (6) in the lowermost capillary compartment. The diffuser uses a plug at the end of the refill line, with many holes machined along the length of the tube. The holes should be uniformly distributed along the length of the tube within the lowermost compartment. Holes size is not too important as long as the size is greater than about 0.25 inch and less than 2 inches. Total hole area should be at least ten times the cross-sectional area, that is 1.96 ft^2, thus reducing the inlet momentum by a factor of about 100. In addition to the effect of the diffuser, the capillary barriers which subdivide the tank into compartments dissipate part of the energy. This technique should allow orderly refueling, although further work needs to be done on diffuser design and the magnitude of capillary barrier dissipation.

Using a refueling rate of 30,000 lbs/hr (10 hours to accomplish refueling), the pressure loss through the valves and the fill line (5) is about 2.2 psi for a 6-inch diameter line. Although a detailed tradeoff on fill line size was not done as part of this phase of the RNS study, such a study was carried out as part of NR's Orbital Propellant Depot contract and a 6-inch line size was recommended. Therefore, as the refueling rate is compatible with the present logistic plan and the pressure loss is acceptable, a 6-inch refueling line is recommended at this time.

Pressurization

The Phase II study showed that a bootstrap pressurization system, called "autogenous pressurization", resulted in appreciable weight saving over other candidate systems. In principle, autogenous pressurization utilizes engine heat to vaporize some of the LH_2 pumped from the tank and thus the hydrogen gas is then returned to the tank maintaining the tank's pressure as the expulsion continues. A turbopump system is used to feed propellant to the engine - with turbine exhaust working fluid, supplied via a by-pass from the engine, being used as pressurant. The start-up phase when engine chamber pressure, pump spin rate, and turbine exhaust supply are building up is the critical operational period. Pressurant supply rate requirements are appreciably higher during this period than during steady engine burn, rate requirements surging to stay ahead of pump requirements.

In performing the pressurization analysis before ANSC's start-up analysis (Reference 5.2) had been completed it was necessary to make some assumptions as to tank pressure requirements. In making this first cut, , it was assumed that tank pressure rise during bootstrapping was linear with time. Detailed results are presented in the Thermodynamic Analysis of Section 4. It was found that the highest pressurant rates were needed during the LOI burn. The pressurant rate requirements were 3.4 lb/sec. and 6.07 lb/sec, respectively. As passive propellant management (stratification) is the recommended approach, this lower flow rate is used in line sizing.

To account for uncertainties as to degree of ullage stratification, flow rate upon which the design was based was increased 33% to 4.5 lb/sec. Equations and design charts for compressible pipe flow with friction are presented in Reference 5.3. The charts present results of simultaneous solution of Equations (1) and (2).

$$\frac{P_2}{P_1} = \frac{v_2}{v_1} \left\{ 1 - \left[\left(\frac{k-1}{2} \right) M_1^2 \right] \left[\left(\frac{v_2}{v_1} \right)^2 - 1 \right] \right\} \tag{1}$$

$$\frac{fL}{D} = \frac{1}{k} \left[\frac{2 + (k-1) M_1^2}{2 M_1^2} \right] \left[1 - \left(\frac{v_1}{v_2} \right)^2 \right] - \left(\frac{k+1}{2k} \right) \ln \left(\frac{v_2}{v_1} \right) \tag{2}$$

where

p is pressure
v is specific volume
k is ratio of specific heats (C_p/C_v)
f is friction factor
L is line length
M is the Mach number

Subscript (1) represents conditions upstream, at the turbine exhaust.

Subscript (2) represents conditions downstream, at the tank pressure regulator.

Conditions at the turbine exhaust are pressure = 650 psia, temperature = 260 degrees R, as determined from Reference 5.2. Using the design chart (Figure 8-7 of Reference 5.3), it was determined that line size 2.25 inches in diameter is required. ANSC (Reference 5.2) recommended a 3.5-inch line diameter. This difference can be readily explained, as ANSC assumed a destratified ullage, while NR's analysis is based on a stratified ullage - with ullage pressure at startup greater than the vapor pressure.

A schematic of the autogenous pressurization subsystem is included in Figure 5-12. A bang-bang control system is used operating off tank pressure. Orifice (12) allow less flow than that required for steady engine burn; thus, this flow is supplemented as required with flow through the larger orifice (13) by activation of the on-off valves (14), as required. Orifice (12) plus (13) are oversized for the maximum flow requirement during pressuriation. As required during pressurization, the flow through (13) is shut-off by valves (14) to give the proper pressure build up.

Pressurant enters the tank through the diffuser (15). This causes flow to enter horizontally and at low velocities to minimize ullage mixing. However, during pressure build-up and steady burn, the pressurant flow rates are 180 and 27 ft^3/sec. These high volumetric flow rates will require a baffle system as well as a diffuser to reduce the incoming momentum of the pressurant. This development effort is regarded as being relatively routine and should be studied in subsequent study phases.

Vent System

The vent system differentiates between boost or ground venting, orbital venting and emergency venting as shown in Figure 5-12.

Ground and Boost Vent Subsystem. The vent line and valves for venting and pressure control during ground loading, ground hold, and boost phases of the mission are denoted by (1) in the figure. Two valves in series are normally closed; they open if tank pressure reaches an incipient critical condition. In its operation, this vent subsystem utilizes the orbital refueling subsystem. The vent line connects to the orbital fill line; thus venting is accomplished via the orbital fill diffuser (6) through the orbital fill line (5) to the vent line (7). Using two valves in series prevents inadvertent venting if one valve fails open. If one valve fails closed (normal position) venting can be accomplished through the orbital fill system (3) and (8). In this case, the shutoff valve (8) and the divider valve (3) are opened to permit full flow through the fill line (5). Both valves open if they sense a pressure of 1 psia greater than the pressure at which valve (1) is programmed to open. For a 6-inch line diameter, the rate of pressure decay due to venting was calculated to be 0.045 psi/sec. This calculation was based upon a vent rate of 52.5 ft^3/sec, based on a conservative evaluation of line friction losses. Substitution of this vent rate into the formula below yielded the pressure decay rate of 0.045 psi/sec.

$$\frac{dP}{dt} = \frac{\frac{C_pMP}{R}\,\frac{dV}{dt}}{\left(\frac{C_pM}{R} - 1\right)V}$$

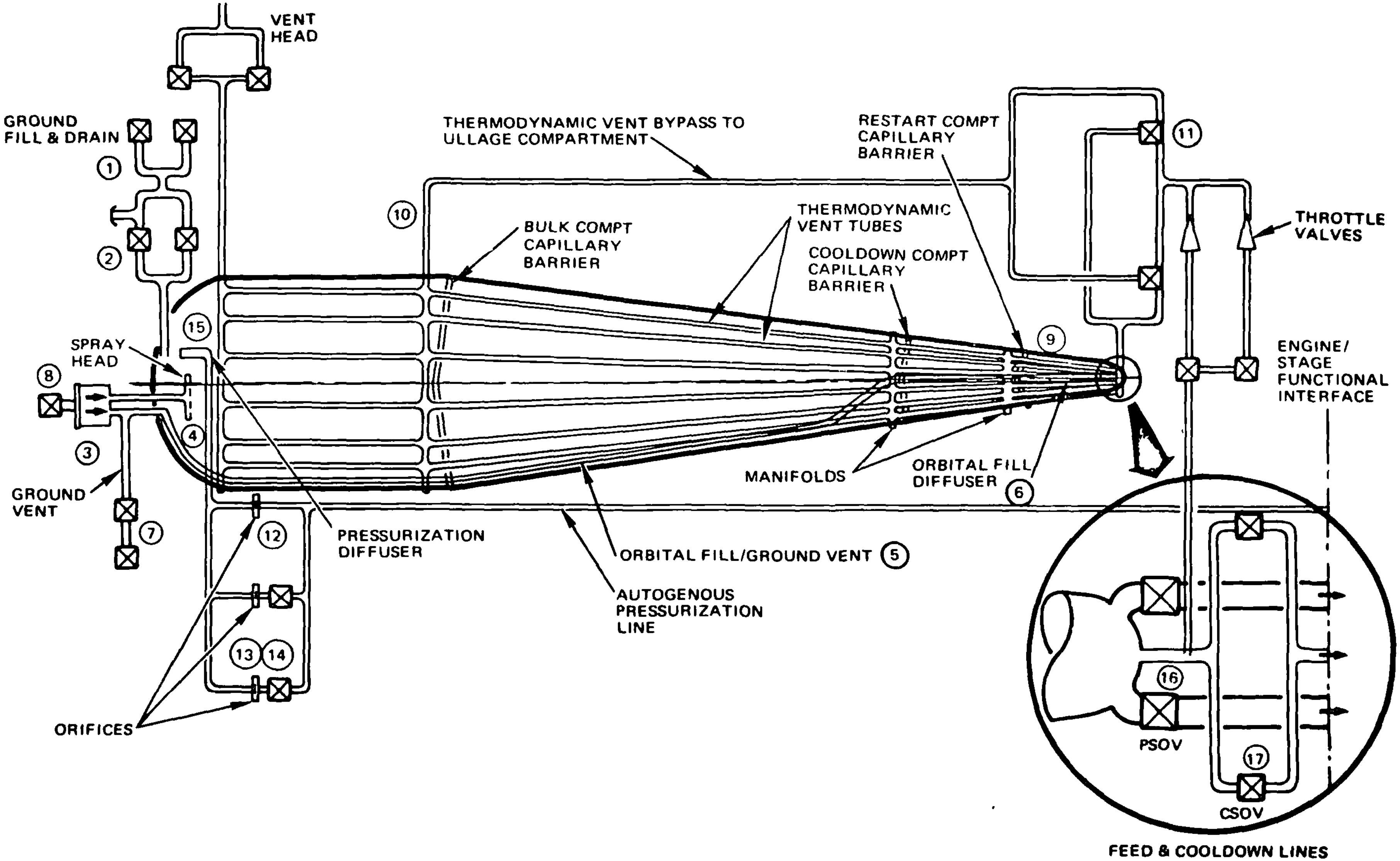

Figure 5-12. Pressurization and Vent System Schematic

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In the above equation, C_p is the specific heat, M the molecular weight R the ideal gas constant, P the tank pressure, and V the tank volume. This equation was derived as part of the Thermodynamic Analysis in Section 4. This vent rate is deemed adequate and as the line size is compatible with that required for orbital refueling, a 6-inch diameter ground and boost vent line is recommended.

Thermodynamic Vent System. This vent system has the dual function of preventing pressure excursions and maintaining subcooled propellant within the capillary compartments. The system, as shown in Figure 5-12 is in effect an open loop refrigeration system and has many features in common with the design being developed by NR for the Space Shuttle orbital maneuvering propulsion system.

The system operates by withdrawing liquid through the cooldown line from the lowest capillary compartment and throttling the liquid to a lower temperature and pressure. The thermodynamic state of the expanded fluid usually corresponds to a two phase (liquid and vapor) condition. The on-off valve upstream of the throttling valves operate off both tank pressure and liquid temperature. Venting is initiated if tank pressure or liquid temperature rises over prescribed values. The throttle pressure adjusts to maintain a sufficient temperature difference for cooling the LH_2. The throttling valves sense the temperature of the liquid within the capillary compartments and the hotter the propellant temperature, the lower the throttle pressure and the lower the vented fluid temperature.

Under nominal conditions the fluid flows to heat exchanger tubes (9) at the bottom of the RNS tank to assure that this important region and the feedout lines receive adequate cooling. The tubes have a hemispherical cross-section and are affixed (probably by welding) to the outer tank wall to assure good thermal contact. The thermodynamic tubes run the length of the tank in parallel and are connected together at various levels by manifolds. Means to isolate a leak in either tube or manifold need to be developed to safeguard that a few leaks will not result in total system failure. The tubes cool the liquid propellant and tank walls, and intercept heat leaks through the high performance insulation. The flow rate is regulated so that the cooling fluid becomes completely vaporized upon reaching the ullage compartment. This is accomplished by flow rate control valves which sense the temperature of the vented fluid, assuring that the vent fluid is substantially superheated.

This system is subject to the override that if tank pressure begins to rise rapidly, cold throttled fluid is by-passed directly to the cooling tubes along the ullage compartment (10). If necessary, all the throttled fluid can be by-passed in this way resulting in rapid cooldown, including possible ullage vapor condensation, of the ullage compartment. The proportioning valves (11) controlling this by-pass operate off tank pressure.

The in-orbit vent system has been designed to have more than adequate capacity for anticipated solar and nuclear heating. This provides ample safety factor and allows utilization of the system for venting during orbital refueling. Analysis of this system proceeded from related studies done at NR for the Space Shuttle. In particular, the nominal pressure of the fluid after throttling was 5 psia. This provides two phase fluid about 6°R cooler than that within the capillary compartments and allows 3 to 4 psi flow loss. The average heat leak is 1280 Btu/hr., which requires a vent rate of 6.74 lb/sec. of saturated vapor or 3.37 lb/sec. of superheated vapor at 130°R. It is anticipated that this higher vent fluid temperature can be obtained; however, the flow rate for sizing of the vent system is considered at 6.74 lb/sec.

The recommended design is 0.5 inch hemispherical cross-section tubes spaced at 3-ft. horizontal intervals along the cylindrical portion of the tank. The tube spacing or number of tubes must, of course, change within the conical portion of the tank. Pressure drop for this configuration was calculated to be 2.7×10^{-3} psi for the nominal vent rate. Thus, the design has flexibility in that low throttle pressure (for greater cooling capability) and can be obtained without compromising vent flow rate capability. In addition, the system has the capability to handle higher vent rates as would occur with on-off vent system operation, higher heat leak rates or venting during refueling.

The in-orbit vent system is fully redundant; each individual system has three valves to assure proper flow shutoff. However, an emergency vent system is provided to handle contingencies.

Emergency Vent. Unanticipated high rates of heating will require an emergency vent provision. No practical system could handle a worst case situation such as a detonation. However, greater vent capability than that provided by the in-orbit system to handle non-catastrophic situations is required. An 8-inch diameter emergency vent line can pass about 45 ft^3/sec of liquid or 360 ft^3/sec of vapor corresponding to pressure relief (for 2/3 empty RNS) rates of 0.039 and 0.312 psi/sec., respectively. This is believed to be adequate for non-catastrophic contingencies and, therefore, an 8-inch line diameter is recommended. This system should be more than adequate for venting during orbital refueling.

AUXILIARY PROPULSION

The RCS selected for the RNS consists of 4-quads mounted 90 degrees apart on the forward skirt of the stage. Each quad consists of two 250-pound axial thrusters and two 25-pound radial thrusters for pitch, yaw and roll control. The reactant employed is O_2/H_2 supercritically stored.

System Design Requirements

The most critical control maneuver demanded for the system is docking with a payload in lunar orbit since a mission abort due to a single point failure at this time would be unacceptable. Also, although the docking maneuver is expected to be automatic, manual docking backup is required. Nevertheless, the maneuver requirements established to date for the lunar shuttle mission permit a relatively leisurely manner of accomplishment. That is, no maneuver needs to be accomplished in a few seconds; several minutes or longer can be taken. Additionally, continuous thrusting for as long as 1000 seconds is considered acceptable. This design approach reduces the accelerations required for any given maneuver and permits employing a smaller size thruster than recommended during the Phase II study. Also, the midcourse corrective maneuvers have shown to be more economical when performed with the NERVA, reducing in half the RCS thrust recommended during the previous phase of the study. Additionally, thrust levels less than 300 pounds are compatible with the capillary devices designed for propellant location control. The flexibility of the devices increase as the thrusters size decreases.

The total RCS impulse requirements for the lunar shuttle mission is 1, 900, 000 lb/sec. This amounts to a total propellant requirement of approximately 5, 800 pounds.

O_2/H_2 Supercritical System

The supercritical storage gaseous O_2/H_2 system is recommended for the Reusable Nuclear Shuttle because the concept exhibits a minimum amount of development problems. The supercritical storage concept for both hydrogen and oxygen has been used in previous spacecraft so valuable development experience is already available. Additionally, this type of system eliminates the problem of propellant acquisition since supercritical storage assures single phase delivery regardless of the gravitational conditions.

The major area needing development is in the required propellant conditioning. This is basically accomplished by heating the propellants in a heat exchanger which is provided energy from the flow of hot gases from an O_2/H_2 gas generator as shown in Figure 5-13. The gas generators, one for each propellant, also supplies the thermal energy to the supercritical storage tanks heat exchangers that are required to maintain the propellant in a supercritical state. This is done by flowing hot propellant from the primary heat exchanger into the storage tank heat exchanger. A bypass flow control loop is utilized to control the propellant temperature out of the primary heat exchanger. The propellant from the storage tank heat exchanger is further conditioned in a secondary heat exchanger which also contains a temperature sensing bypass flow control loop. The final product is then stored in accumulators whose size is determined by pressure limits, propellant temperature and the system cycle requirements. From the accumulators, the propellants flow through pressure regulators and then to the engines.

Because of the heavy storage tanks and number of components, this system weighs more than other candidates considered. However, due to the high total impulse requirements for the RNS, the system hardware weight penalty has minimal impact on the selection.

Figure 5-14 shows the schematic of the system depicting a fail operational/fail safe capability.

ASTRIONICS

The astrionics bay shown in Figure 5-2 contains all the electronic equipment including guidance, navigation, controls, communication, and the NERVA NDIC. In addition RCS, electrical power system, and environmental control are housed in this area.

The distribution of the equipment both within and without the astrionics bay is displayed in Figure 5-15. All instrumentation and components identified during the course of the study are shown on the developed views of the inner and outer surfaces of the astrionic module structure.

The majority of the inside surface is covered by the two RCS propellant tankage quadrants with a third quadrant occupied mainly by three fuel cells with their associated inverters and controls. Also shown are the two batteries required during peak NERVA operations. The NDIC is located next to the electrical power system. The last internal quadrant is used for

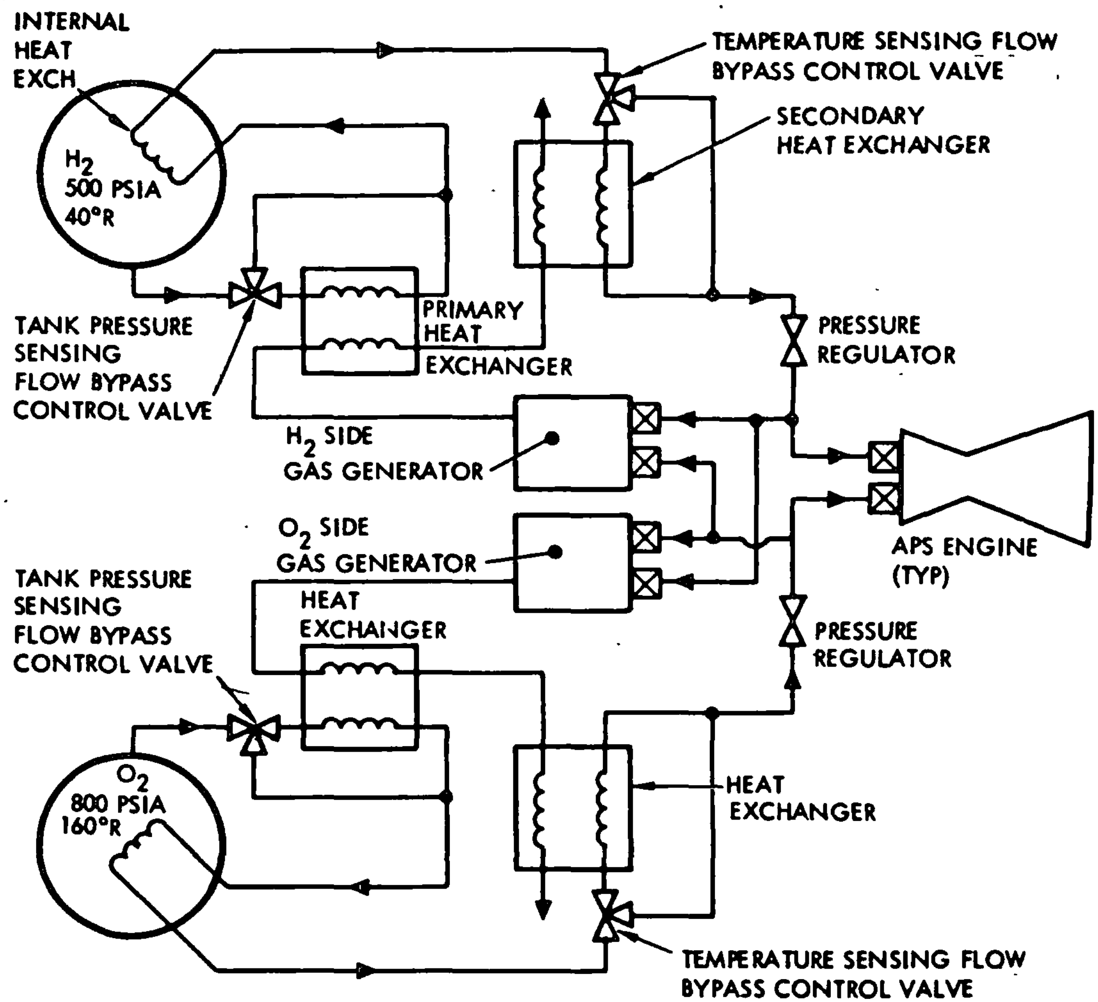

Figure 5-13. O_2/H_2 RCS - Supercritical System

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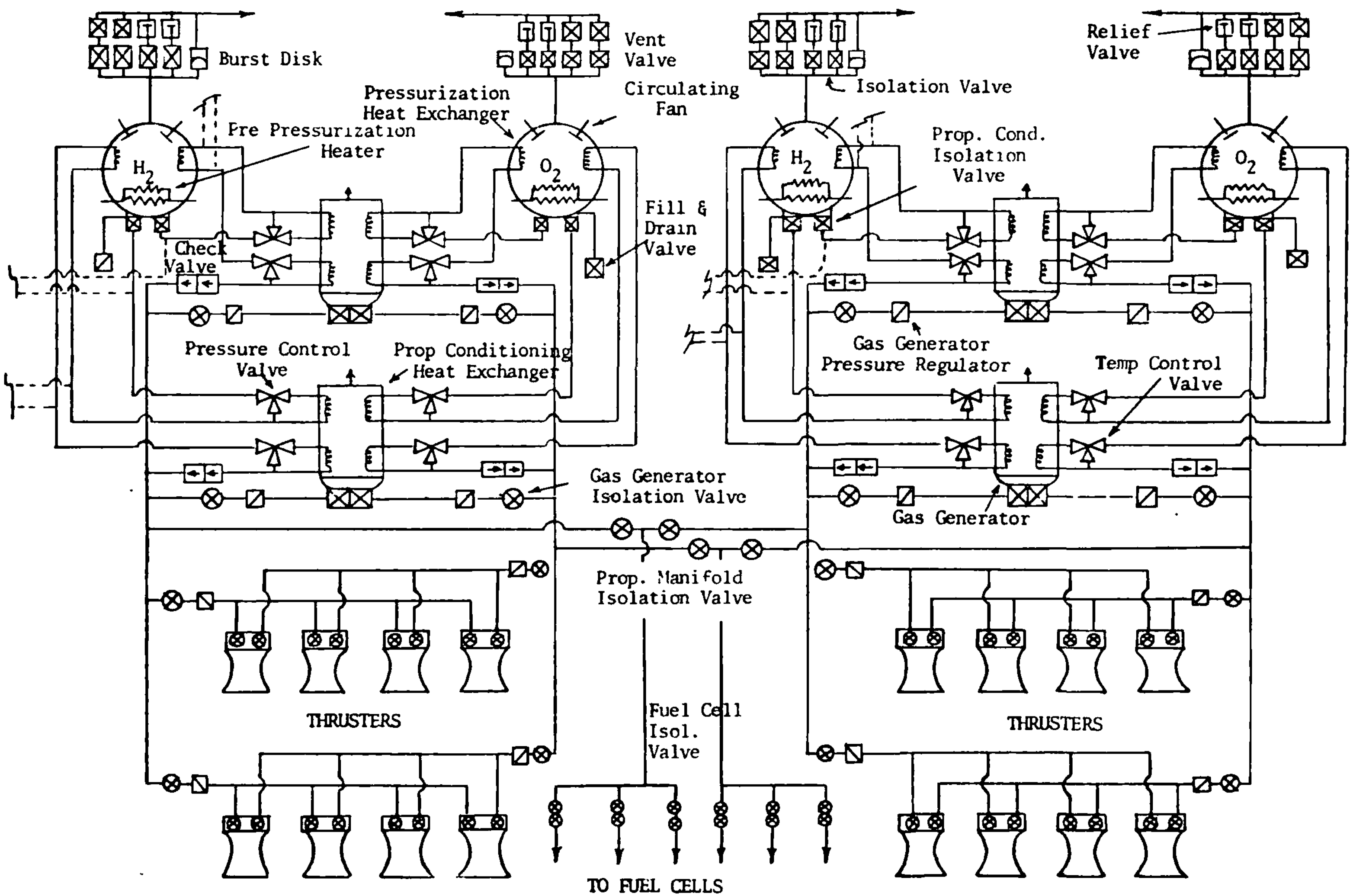

Note: **Dotted Lines Indicate Connections to Other H_2 Tanks**

Figure 5-14. RNS-Reaction Control System

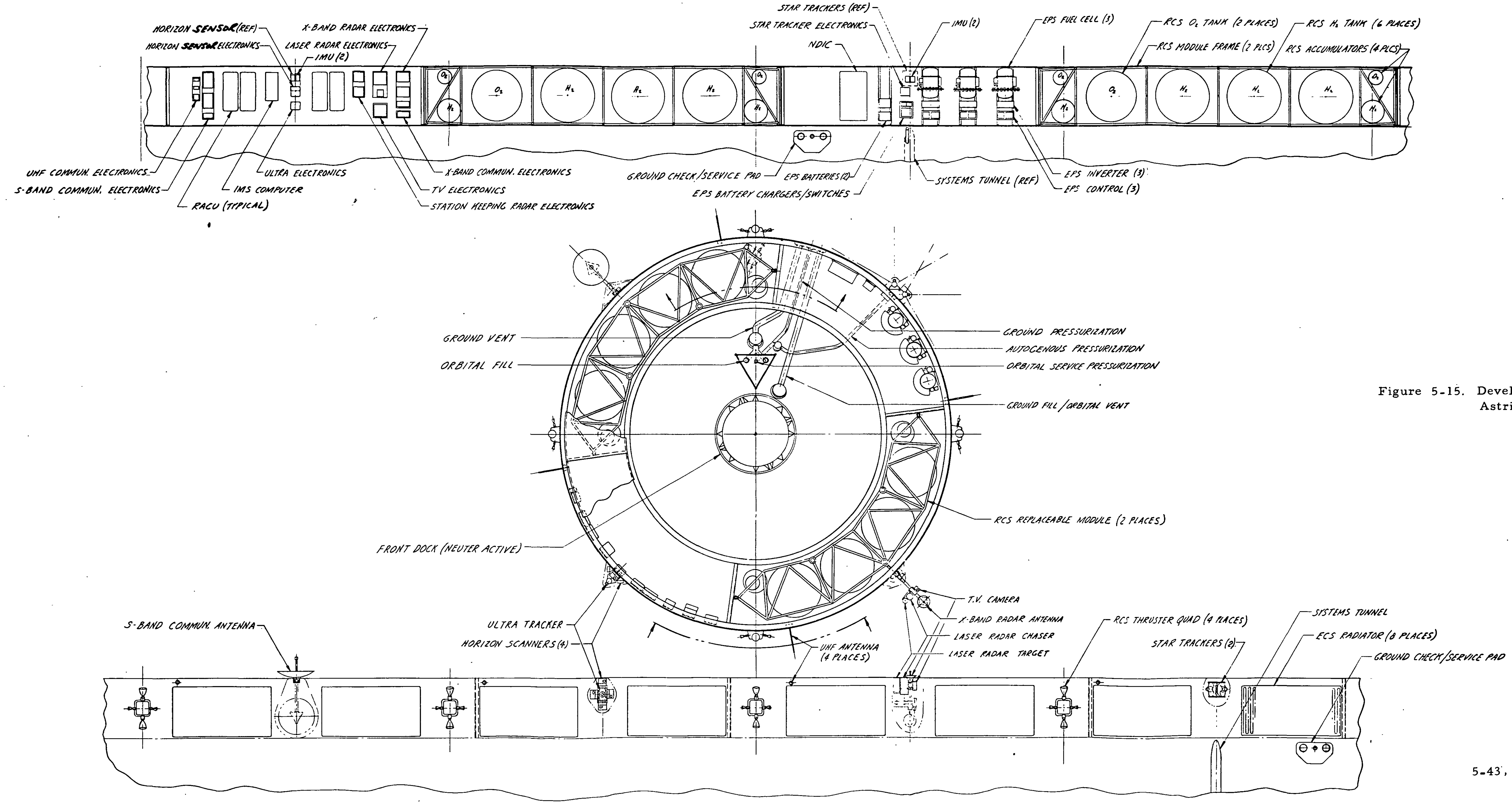

Figure 5-15. Developed Views of the Astrionic Bay

the instrumentation connected with horizon sensors, communication, station keeping and docking, IMS computer and RACU's. Four inertia measuring units are used to satisfy the fail operational/fail operational/fail safe criteria employed in the definition of electrical and electronic subsystems.

The external surface of the astrionics bay is mostly covered with the environmental control system radiators and by the four RCS pods. In addition, the star trackers, the communication antenna, the ULTRA tracker and four horizon sensors and the rendezvous and docking equipment consisting of TV camera, laser radar chaser and target and the X-band radar antenna occupy the remaining available space.

As can be seen, the surface and volume occupied by the electronics and equipment, other than the RCS and its propellant requirement, is relatively small. Their relocation to suit additional or new refurbishment requirements can be accommodated.

An autonomous system has been defined for the RNS to provide maximum flexibility for carrying out lunar/geosynchronous and orbit shuttle missions. The capability minimizes the need of earth based operations other than for monitoring, flight scheduling, and resupply. Another obvious advantage to providing the RNS with a completely autonomous capability is that redundant equipment is provided for back-up by the MSFN during orbit-to-orbit transfers, including trans-earth and translunar flights, and by both the MSFN and another autonomous vehicle during rendezvous and docking.

Automatic rendezvous and docking with manual override is consistent with the autonomous capability requirements. Also, there are some obvious disadvantages to requiring a man-in-the-loop for rendezvous and docking. Visual cues would have to be provided either by having a man on board one of the vehicles or by having TV. If the man were on the earth or Space Station for example, time lags introduced by data relay satellite systems could present a problem.

Geometrical and error sensitivity considerations have been used to establish mission requirements for phasing orbit accuracies prior to a rendezvous transfer. These mission requirements translate into constraints which the Guidance, Navigation and Control must satisfy. For phasing orbits these constraints translate into establishing the orbit with sufficient accuracy to allow the RNS radar to detect a target vehicle with a 99.9% probability at a desired detection range and within prescribed scan limits, so that the root-sum-square of delta-V's to correct cross range altitude position errors and velocity errors in all three axes is less (within a 99.7% probability) than could be corrected by a delta-V of the same magnitude as that required for the rendezvous Hohmann transfer.

Using the phasing orbit as a baseline orbit, the Guidance; Navigation, and Control is then required to accomplish translunar, trans-earth, and geosynchronous orbit transfers so that injection into the new trajectory results in the RNS entering desired arrival corridor with a midcourse correction of 50 fps or less. Finally, the phasing orbit requirements must be met following insertion into a new orbit after these transfers.

The Information Management System (IMS) investigated for these operations include digital computer(s), and communication and tracking subsystems to provide the major functions of data handling, control, and communications. The functions are integrated into a system by digital computer software with provisions for crew participation during manned operations. The central component of the system is the digital computer which provides Guidance, Navigation and Control computations as well as communication signal routing, mission planning and event scheduling, on-board checkout, monitor and alarm logic, and operations data management.

A functional description of the selected Guidance, Navigation, and Control components is presented below encompassing an integrated definition of the system. Also, a detail component breakdown is presented in the Mass Characteristics Section following the NASA established weight reporting format.

Inertial Measuring Unit (IMU)

Sensors for establishing inertial state vector, or at least components of the state vector including vehicle linear velocity, linear position, and altitude are commonly referred to as the IMU. In practice, the sensors output are transmitted to the on-board digital computer where actual altitude velocity, and position information is maintained. A strapdown IMU is recommended for the RNS as a result of a detailed NR/SD trade study. The strapdown comparison performed resulted in the MICRON system being the preferred IMU candidate. MICRON, which is the acronym for "micro navigator", is a lightweight, low cost system that incorporates large scale integrated electronic technologies. The main constituents of the MICRON are two small electrostatically suspended gyros (ESG's), a miniature 4096-word/24-bit metal oxide semiconductor computer, servo electronics, power supply and battery, environmental control, and structural housing. The system weighs three pounds, occupies less than 90-cubic inches and consumes about 12 watts of power. The gyro has only one moving part, a solid ball, supported electrostatically. The micron-ESG can serve as an accelerometer as well as a gyro; therefore, two instruments per system provide complete information. The total number of parts used in assembling the micro-ESG is approximately

40 as compared to 250 or more for an inertial grade gyro in a more conventional system.

Existing MICRON systems exhibit a 0.04 deg/hr rms random drift rate and a 2,000 hour MTBF. Projected capabilities of the space-qualified instrument are a 0.01 deg/hr and 16,000 hours, respectively. To provide a high reliability, use of four MICRON systems per vehicle is anticipated. Present accuracy of the MICRON acceleration sensing capability is in the order of 10^{-4}. Six additional accelerometers are included in the power, weight, and volume to cover any requirements for greater accuracy.

Inertial State Vector Updating Sensors

Inertial state vector updating sensors can be subdivided into two classes - attitude and position/velocity sensors.

Attitude Updating Sensors

Vehicle inertial attitude can be determined by several methods. Determining line-of-sight to two start was selected for the RNS because of its simplicity, accuracy, and applicability to all phases of the reference lunar mission. Mechanically gimballed star trackers were selected for the attitude updating. These instruments have the capability of tracking stars within a field of view (FOV) determined by the gimbal freedom. Star acquisition is accomplished by scanning a portion of the celestial sphere in which a known star is expected. Since gimbaling can provide a large FOV, these instruments are, in general, less constrained operationally than other types evaluated. For further details, please refer to Volume II - Part A - Section 4 of this report.

Position Updating Sensors

Accurate space navigation has here-to-fore utilized ground based tracking networks for vehicle position updating. The requirement for accurate automatic autonomous navigation presents significant problems which have not yet been fully investigated. These include the determination of what observable phenomena yield navigation data. Observable phenomena include the following:

- Angles between the lines of sight to selected celestial objects
- Subtended angle of planet disk
- Curvature of a planet horizon

- o Time of known star occultations
- o Ranging measurements to planets or navigation satellites
- o Refraction angle of a star near a planet disk
- o Landmarks
- o Ions in the upper atmosphere

Observation of one or more of these phenomena lead to measurements used to update the estimate of the vehicle state which is maintained continuously in the on-board digital computer using IMU outputs.

The study of position updating sensors is complicated by the fact that less information is readily available than for the IMU. However, a number of qualitative judgements are possible and are the basis of a preliminary selection of a horizon sensor, and an unknown landmark tracker (ULTRA) for state vector position updating.

Horizon Sensor. A horizon sensor in general terms is an instrument to detect the limb of a planet or moon. For earth operation present horizon sensors detect energy in the 14-16 micron CO_2 absorbtion band. This band is also suitable for detecting the hot side of the moon which radiates over a broad band. However, it is not satisfactory for cold side of the moon operation where approximately 90% of the radiation is above 13 microns. The 22-40 micron region is suitable for both hot and cold side of the moon operation and furthermore includes a water vapor absorption line suitable for earth sensing. Horizon sensors used for RNS applications must cover a large range of subtended angles. Discussions with Quantic Industries have indicated the feasibility of having two optical assemblies in each sensor head. One would operate at altitudes in the 80 to 15,000 n.mi. region and the other at higher altitudes. Power, weight, and volume horizon sensor estimates given in Table 5-8 are for this dual optical assembly Quantic Industries 4-head, fully redundant system.

Table 5-8. Horizon Sensor Characteristics

Characteristics / Component	Power (Watts)	Weight (lb)	Volume (in^3)	Accuracy (1σ)
Tracker Heads	-	-	720	
Electronics	-	-	510	
Total	38	49	1230	2 min

However, even if the horizon sensor hardware were perfect, errors would still be present in orbit because the earth's IR radiation is not constant. Errors due to changes in earth radiation are not the same in all kinds of horizon sensors; each concept is subject to a different amount of error, depending on the way it finds the horizon. This error can be calculated for any given sensor employing earth's horizon data and the sensor specifications. Any portion of the error not calculated and not of short term duration becomes a bias error. Horizon sensing bias errors not accounted for in the error calculation translate into navigation errors. The correlation between unaccounted for bias errors and navigational accuracy is recommended to be investigated in future RNS development efforts employing the Mission Accuracy Requirements Computer Program (MARC).

Unknown Landmark Tracking. Unknown landmark tracking has been developed by the Autonetics Division of NR and has been proven feasible by deterministic methods similar to LaPlace orbit determination used by astonomers (the problem is actually reversed since the tracking involves looking at earth from orbit, but this actually reduces the diffraction induced errors which have plagued astonomers). After feasibility was established using deterministic methods, the technique was mechanized using statistical techniques (Kalman filter modified to account for random walk errors).

The tracker uses two narrow photosensors to scan a field of view of 240 arc seconds by 240 arc seconds (about 0.1 miles x 0.1 miles from a 100 mile orbit). The scan is performed first in the X direction, then in the Y direction, then in the X direction and so-on. The photosensor output is subsequently subjected to a threshold, digitized, and converted to a 60-bit word.

An Autonetics patented line-line-scan correlation method is used to produce a correlagram of the correlation from one scan to the next. The landmark is rejected if the intensity pattern does not meet certain specifications (to avoid tracking bright clouds) or if the correlagram is not adequate for tracking (a narrow unimodal shape is preferred). The scan is repeated at one second intervals and the location of the maximum correlation is used to measure the relative motion of the landmark and track it. In-plan landmarks are preferred and typically 3 to 5 different landmarks can be tracked per orbit (as determined by Monte Carlo simulations). It is not practical to perform tracking on the night side of the earth because of the lack of bright light sources or over the ocean because of lack of landmarks.

Autonetics has built a model tracker and has flown test flights in a small airplane at 6000 feet with scaled optics. The performance accuracy is classified but it is obviously better than star horizon measurements alone and not as good as automatic known landmark tracking. The performance is degraded by the inability to obtain suitable night side landmarks. The use of infrared tracking for the night side does not seem practical for this type of measurement because of the associated instrument errors.

A problem exists in that one output of the Kalman filter diverges after a long period of tracking. A technique for truncating old data and initializing the covariance matrix has been developed which allows retention of some of the old data but this has not been implemented. A flyable ULTRA system could be available in about one year if space rated gimbal bearings with the proper accuracy and preload characteristics are available. The device will weight 10-15 lbs and use 25 watts. Volume is estimated at 1230 cubic inches.

The ULTRA output can be used to provide good ephemeris data in earth or lunar orbit, but is not effective at altitudes above 1000 n.mi. because of optical resolution limitations. Thus, it complements the capability of the horizon sensor. A disadvantage is that the unknown landmark tracker requires two landmarks approximately 90 degrees apart on the earth or lunar surface to, in effect negate uncertainties in the unknown landmark's elevation.

It is of importance to note that, Class II and III missions need further evaluation to determine if the star trackers are also satisfactory planet trackers and what reliance should be placed on the Deep Space Tracking Network.

Relative Position Sensors

Rendezvous, station keeping, and docking radar and/or lasers determine the relationship of the RNS relative to another vehicle. During rendezvous the components of interest are vector range, scalar range rate, and perhaps line-of-sight rate. During station keeping, vector range is sufficient information. Automatic docking requires vector range, scalar range rate, and relative angular rates. If a man in the loop accomplishes docking, TV and/or visual information to accomplish lateral maneuvers are required in addition to scalar range and range rate.

The relative state vector information is provided at a high data rate and is maintained in the IMS digital computer independently of orbit data provided by the IMU and navigation state vector updating sensors.

Rendezvous and Docking Sensors. In the requirements analysis a radar with an 0.6 degree angle track accuracy was specified. The beam width should be 6 degrees or less to provide this capability, which implies an acquisition and tracking radar antenna 30 cm or more in diameter. The radar was sized for a 30 cm antenna using standard scaling laws.

The scanning laser radar (SLR) package provides chaser and target vehicle capability for the RNS with a 5 to 10 KBPS data rate communication system which can be used as a redundant data channeled for rendezvous and docking. Characteristics of the SLR are presented in Table 5-9.

Station Keeping Sensors. Depending on PD, EOSS, and LOSS traffic patterns and whether a rotational or translational "g" field, or combination thereof is used for propellant transfer an essentially omnidirectional station keeping radar may be required. Four X-band dipoles mounted on the forward RNS structure would provide nearly 4 steradian coverage. Information from any three of the dipoles fixes a target vehicle's vector position. The acquisition and tracking radar or SLR could provide vector range information over a more limited solid angle, but could only position targets within the respective beam widths. The station keeping radar, on the other hand, can fix all targets separable in range and not in the region along the RNS axis shadowed from the station keeping radar. Characteristics of the station keeping radar, and the X-band radar are included in Table 5-9.

Information Management System

The Information Management System (IMS) includes digital computer(s), and communication and tracking subsystems. A multiprocessor computer system was selected for the RNS because of the maximum flexibility it allows, the expansion capability both for added processing and growth to planetary mission capability, and its lower size, weight, and power. Additionally, a multiprocessing system can be developed in parallel with the other subsystems because of its capability to adapt to changes in the subsystem requirements.

The Goddard Tracking and Data Relay Satellite (TDRS) system appears attractive for providing continuous communications. However, preliminary calculations indicate that a 3 to 13-foot antenna, depending on hardware (pre-amp, etc.), characteristics would be required to compensate for a TDRS nondirectional antenna link with the RNS. Direct links with earth would require much smaller antenna size and/or transmitter power, but provide only intermittent coverage in low earth orbit. Since no requirement for continuous communication has been identified, a system

Table 5-9. Acquisition and Tracking Radar, Scanning Laser Radar, and Station Keeping Radar Characteristics

SYSTEM	POWER (Watts)	WEIGHT (lb)	VOLUME (in^3)
X-band Radar			
2 Transmitters (1 watt avg power per transmitter)	8	4.4	640
Receiver	12	5.0	290
Antenna (1.2 ft dia)	30	12.0	300
Totals	50	21.4	1230
SLR			
Target	20		
Sensor		20.0	755
Electronics		6.0	300
Chaser	35		
Sensor		20.0	855
Electronics		9.0	1440
Totals	55	55.0	3350
Station Keeping Radar			
2 Transmitters	8	4.4	640
4 Receivers	50	16.0	920
4 Omni Directional Antennas	-	.5	30
Cabling	-	12.0	90
Total	58	32.9	1680

transmitting intermittently, while in contact with the MSFN, was selected. In the event that MSFN S-band channels are allocated, K-band frequencies at windows in the oxygen and water vapor absorption spectrums appear attractive. The UHF system provides omnidirectional vehicle-to-vehicle, EVA, and short range ground-RNS communications.

The X-band radar provides for rendezvous and may be time shared for vehicle-to-vehicle communications. Operating with a cooperative target, the radar, with a 2-watt transmitter and 30-cm diameter parabolic antenna, has a detection range greater than the 820 n mi requirement for geosynchronous altitude autonomous rendezvous. The scanning laser radar accomplishes automatic docking and also provides for tracking a cooperative target at ranges up to 75 n mi.

Digital Computer

The heart of the information management system (IMS) is the digital computer. It provides for all data handling, processing, and storage on-board the RNS. The requirements used for the digital computer trade study analysis are outlined in Volume II - Part A - Section 4 of this report. General purpose processors have been selected for the RNS because of the maximum flexibility they afford, their growth capability to provide for planetary missions, and their saving in size, weight, and power.

In the comparison that followed between a single centralized processor, several dedicated processors, and multiprocessing system, the latter was chosen because of the maximum flexibility it allows; the expansion capability both for added processing and planetary growth; and its lower size, weight, and power. Additionally, multiprocessing systems can be developed in parallel with the other subsystems because of their capability to adapt to changes in the subsystem requirements. Several recent studies have investigated multiprocessor digital computer configurations in considerable detail. Reference 5.4 gives a survey of existing systems and presents scaling laws. Reference 5.5 discusses preprocessing by local processors, data bus design, and packaging concepts. Weight and volume estimates for the multiprocessor are from Reference 5.5. Power was computed as 50 watts for the first CPU plus 30 percent increase for each additional CPU. Memory power was allotted six watts for each 4,000 32-bit words.

Communications and Tracking

External communication links associated with the RNS fall into the following categories.

1. Communications with ground via direct or relay satellite system links while the RNS is in earth orbit at or below geosynchronous altitudes.

2. Communications with ground while in earth and lunar orbits above geosynchronous altitudes including translunar trajectories and lunar orbits.

3. Communications with earth during interplanetary and circumplanetary flight.

4. Communications with logistics vehicles, Space Station, LOSS, and PD/M&R via direct links.

Providing for communications in the fourth of the preceding categories is straightforward. The third category will use the deep space tracking network. Providing common equipment for the first two categories is more of a problem because of the number of options available. A communications system summary is shown in Table 5-10. The trade-off analysis performed leading to the selection of each component is discussed in detail in Volume II - Part A - Section 4 of this report.

An RF system using MSFN facilities was selected as most attractive on the basis of the qualitative criteria. In the event that MSFC S-band channels are allocated, K-band frequencies at windows in the oxygen and water vapor absorption spectrums appear attractive.

Vehicle-to-vehicle communications are provided by X-band, SLR, and UHF systems. The X-band and SLR systems, also used for tracking, are discussed in the section on relative position sensors. The UHF system provides omnidirectional vehicle-to-vehicle, EVA, and ground RNS communication. Four UHF antennas are provided for redundancy. Equipment characteristics are presented in Table 5-11, including power, weight, and volume.

The schematic diagram shown in Figure 5-16 depicts the main information flow paths among the assemblies which comprise the Navigation, Guidance and Control subsystem. All of the navigation subassembly inputs and outputs can be addressed only through the RACU's. Activation of a guidance program determines which assembly or assemblies will receive commands from the multiprocessing computer or provide data to the computer to satisfy the requirements of the particular phase of the mission.

Table 5-10. Communications System Summary

Communications System Component	Communications Purpose	Comments
X-band system	Inter-vehicle	Antenna shared with acquisition and tracking radar system.
Scanning radar laser	Inter-vehicle	Provides back-up rendezvous and docking communications capability.
UHF system	Inter-vehicle (short range), ground	Omnidirectional antennas provide for inter-vehicle, ground and/or EVA communication.
S-band	Ground	Data dump from low earth orbit using MSFN facilities. Continuous communication capability with deep space network in geosynchronous, trans-lunar, lunar orbits.

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Table 5-11. Communications Equipment Characteristics

System	Power (Watts)	Weight (lb)	Volume (in^3)	Comments
S-band				
2 transmitters	80	8.0	1300	3-foot diameter parabolic antenna.
1 receiver	15	5.0	300	12-watts transmitter power.
4 omni antennas	-	.5	10	
1 directional antenna	30	19.5	7300	
cabling	-	12.0	90	
Totals	125	44.5	9000	
X-band				
electronics	15	5	350	Electronics (encoder, etc.) for communications, only. Other equipment (antenna, etc.) covered under relative position sensors.
SLR				All equipment covered under relative position sensors.
UHF				
2 transmitters	6	1.0	70	
1 receiver	4	2.0	100	
4 antennas		.5	30	
cabling		12.0	90	
Totals	10	15.5	290	

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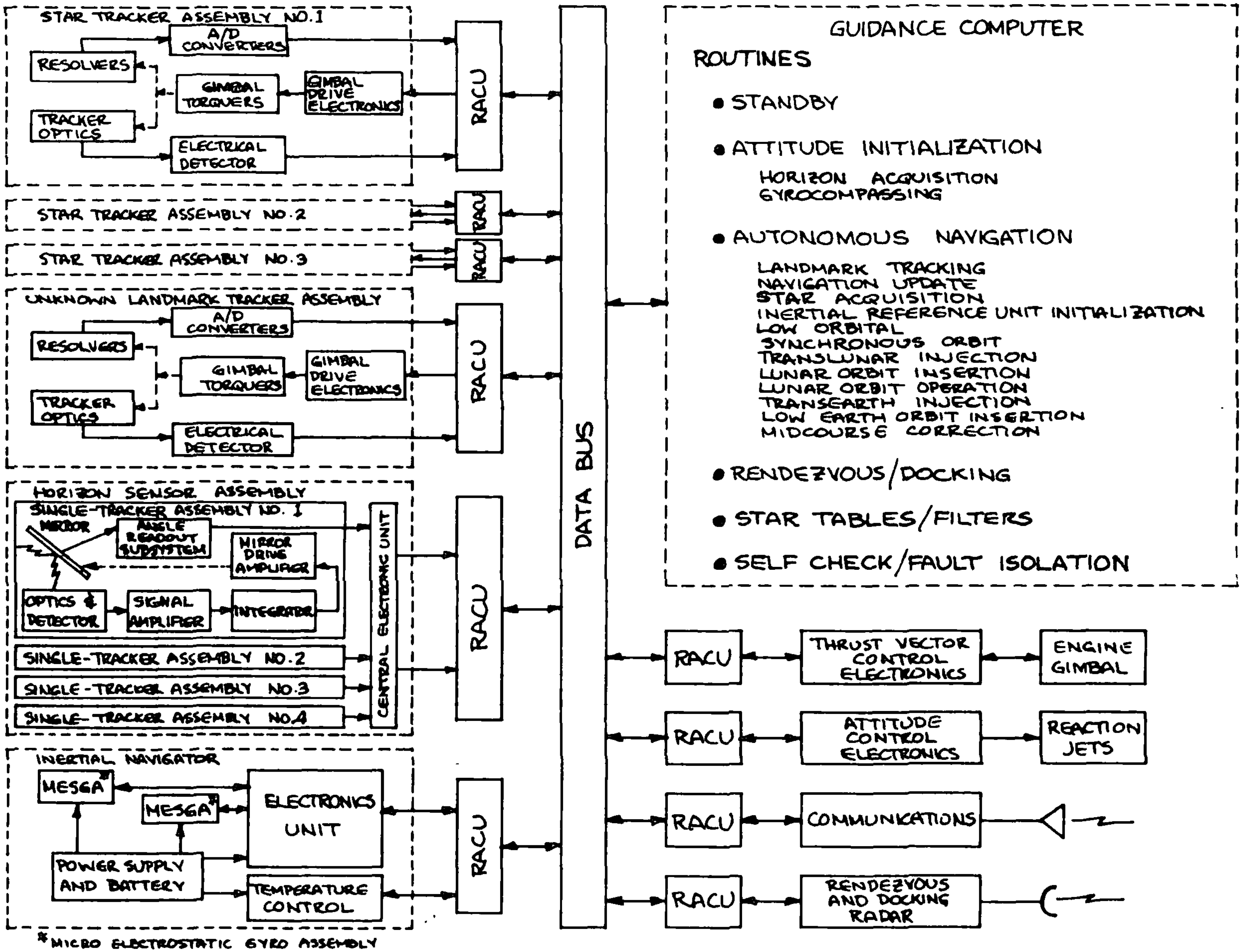

Figure 5-16. Guidance, Navigation, and Control Block Diagram

Electrical Power

The RNS self-sufficing concept requires that all electrical energy requirements be provided from on-board equipment during each mission cycle. Power cycles of 28 and 55 days have been investigated with a minimum stage lifetime of three years. Electrical power during dormant periods other than those in the mission cycle is assumed to be supplied from the propellant depot and/or other power modules. The subsystem concept and component selection is based on maximizing commonality with other space program elements (i. e., Space Shuttle, Space Station, etc.) and minimizing special logistic requirements, where practical.

The redundancy requirements are dictated by the mission including considerations for manned payload missions. Three separate and independent busses are to be provided to satisfy redundancy requirements. Automatic circuitry is to be provided to permit paralleling of energy sources, inverters and/or converters with minimum transient effects and for load protection. Both back-up and emergency power requirements are considered in the final concept selection.

Requirements

ANSC report S-130, "NERVA Engine Reference Data," September 1970, Contract SNP-1 was utilized as the source document in identifying the NERVA power requirements. The engine firing can be divided into three distinct periods from a power consumption viewpoint: (1) engine operation including startup, steady-state, and shutdown; (2) cooldown; and (3) coast.

The power requirements for a typical engine firing include operating engine valves and control drum actuators as well as the engine control and instrumentation power. This value, when averaged over the engine operating period, is approximately 2, 200 watts. Peak power during this time may be 3, 500 watts at initiation of chilldown.

The power requirements during cooldown and coast are much lower than during engine operation. Power requirement during cooldown is about 200 watts with additional 400 watts pulses, which represent valve operation. However, because the valves used during pulse cooldown require no power except when actuated, valve operation adds only about one watt of continuous power. The coast power requirements (approximately 60 watts) are based upon the use of a limited amount of critical instrumentation and maintenance of the command receiving capability.

Although the power requirements are lower during cooldown, the amount of energy required is much greater than for engine operation as shown in Table 5-12.

Surge currents will be significant only for the actuators. Initiation of chilldown will draw as much as 128 amp for approximately five seconds during normal operation. Other surges will be 70 amp or less for a period not exceeding five seconds.

The stage-mounted digital electronics, located in the forward skirt area of the nuclear stage, will dissipate 97 percent of the NERVA electrical energy during the complete mission. The remaining three percent will be dissipated in the engine area for valves and actuators.

Power requirement for the engine during checkout is 200 watts and is presently identified to have a duration of 600 seconds. This power level is considerably lower than the operating power level previously assumed for the checkout function based on earlier data.

The astrionics electrical power requirements listed in Volume II - Part A - Section 4 of this report are the latest estimates based on definition of various components identified in the Phase III study. The electrical networks power requirements are those developed during the Phase II study. The NERVA engine power levels shown in Table 5-13 are the exact power required for each NERVA engine operation.

The electrical energy requirements identified in Table 5-13 are based on the "NR Nominal" lunar missions. The stay time in lunar orbit is approximately 17 days, 13 of which the RNS is assumed to be in a dormant mode. Since the minimum energy transfer opportunities occur every 54.6 days, it has been assumed the RNS will be in a dormant condition for 24 days in earth orbit, allowing time for stage checkout, rendezvous with the propellant depot, and transfer of propellants.

Energy requirement of NR's representative lunar mission from time of Earth departure to return to Earth operations orbit is shown in a subtotal (777.4 kwh) in Table 5-13. The total energy requirement for a 54.6 day turnaround time is 1196.41 kwh. It should be noted that the long cooldown time and the long dormant times are the major contributors to the overall electrical energy requirement even though the power levels are modest.

Unofficial information received from ANSC could influence the sizing of the RNS electrical power source in the future. That is, ANSC is considering the use of a battery located below the NERVA/stage interface point to reduce

Table 5-12. RNS Subsystem Power Requirements

RNS Subsystem (1) 200 Watts for 600 Seconds for checkout (2) Engine Run Time Dependent- 3500 watts Peak at Engine Start, 2200 Watts during Engine Steady State Run	CHECKOUT		STATION KEEP	MAIN BURN	COOLDOWN	COAST ENROUTE	RENDEZVOUS	DORMANT	REFUEL
	INT.	CONT.	CONT.	CONT.	CONT.	CONT.	CONT.	CONT.	CONT.
Engine	(1) 200	-	60	(2)	200	60	60	60	60
Propellent Gaging	240	-	-	240	240	-	-	-	240
Pressurization	-	160	160	160	160	160	160	160	160
Electrical Control	-	100	100	100	100	30	100	30	30
Environmental Control	-	290	290	290	290	290	290	290	290
Reaction Control	10	10	10	10	10	-	10	-	-
IMU	-	136	136	136	136	139	136	-	-
Star Tracker (2)	60	-	60	60	60	60	60	-	-
Horizon Sensors	38	-	38	38	38	-	38	-	-
Radar X-Band	50	-	-	-	-	-	50	-	-
Radar SLR	55	-	-	-	-	-	55	-	-
Radar Station Keep	58	-	58	-	-	-	-	-	-
Computer	-	310	310	310	310	310	310	-	310
Communications X-Band	15	-	-	-	-	-	15	-	-
S-Band	-	125	125	125	125	125	125	125	125
UHF	10	-	10	-	-	-	10	-	-
Television	100	-	-	-	-	-	100	-	-
T/M RACUS	-	50	50	50	50	50	50	-	50
T/M Signal Cond	-	20	20	20	20	20	20	-	20
T/M Transducers	-	10	10	10	10	10	10	-	10
Total	838	1211	1437	1549	1749	1254	1599	665	1295

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Table 5-13. RNS Electrical Energy Requirements
(Reference Lunar Shuttle Mission)

Mission Phase ① 3500 watts peak at engine start, 2200 watts during steady state engine run.	Mission Phase Duration (Hours)	Power Level (kw) Eng. Burn	Power Level (kw) Stage	Energy Level (kwh) Eng. Burn	Energy Level (kwh) Stage
Payload Mate (Station Keep)	25.50		1.437		36.70
Checkout RNS/PL (Station Keep)	0.5	--	1.437		0.72
TLI 1 Burn (950 Sec SS)*(1620 sec PTO)**	0.45	①	1.549	1.14	0.70
TLI 1 Ellipse Cooldown	4.0	--	1.749		7.0
TLI 2 Burn (770 Sec SS)(1370 sec PTO)	0.38	①	1.549	0.98	0.59
TLI2 Cooldown	10.0	--	1.749		17.49
Midcourse Correction (0 sec SS) (520 Sec PTO)	0.144	①	1.549	0.41	0.22
TLI2 Cooldown Cont.	82.	--	1.749		143.5
LOI Burn (320 sec SS) (710 sec PTO)	0.197	①	1.549	0.54	0.31
LOI Cooldown	66.7	--	1.749		115.2
Rendezvous to Standoff Posit. (RENDZ)	2.0	--	1.599		3.2
Dormant	323.0	--	0.665		215.0
Rendezvous to Station Keep Posit.	2.0	--	1.599		3.2
Station Keep (PL MATE)	25.5	--	1.437		36.7
Checkout RNS/PL (Station Keep)	0.5	--	1.437		0.72
TEI1 Burn (100 Sec SS)(370 Sec PTO)	0.103	①	1.549	0.30	0.16
TEI1 Cooldown	12.0	--	1.749		21.0
TEI2 Burn (0 Sec SS)(190 Sec PTO)	0.053	①	1.549	0.18	0.08
TEI2 Coast	12.0	--	1.254		15.1
TEI3 Burn (30 Sec SS)(260 Sec PTO)	0.072	①	1.549	0.23	0.11
TEI3 Cooldown	9.7	--	1.749		17.0
TE Coast	0.3	--	1.254		0.37
Midcourse Correction (0 Sec SS)(250 Sec PTO)	0.069	①	1.549	0.22	0.11
TE Coast	73.0	--	1.254		91.6
EOI Burn (500 Sec SS)(990 Sec PTO)	0.275	①	1.549	0.72	0.43
EOI Cooldown	24.0	--	1.749		42.0
Hohmann Transfer & Dock - Cooldown Cont.	2.0	--	1.749		3.5
Subtotal				4.72	772.71
Subtotal - Engine plus stage				777.43	
Checkout RNS	2.0	--	1.4		2.8
Dormant	576.0	--	0.665		384.0
Checkout RNS	2.0	--	1.4		2.8
Rendezvous with Propellant Depot	2.0	--	1.599		3.2
Transfer Propellant	18.0	--	1.295		23.3
Station Keep (P/L Mate Posit.)	2.0	--	1.437		2.88
Total					1191.69
Total - Engine plus stage				1196.41	

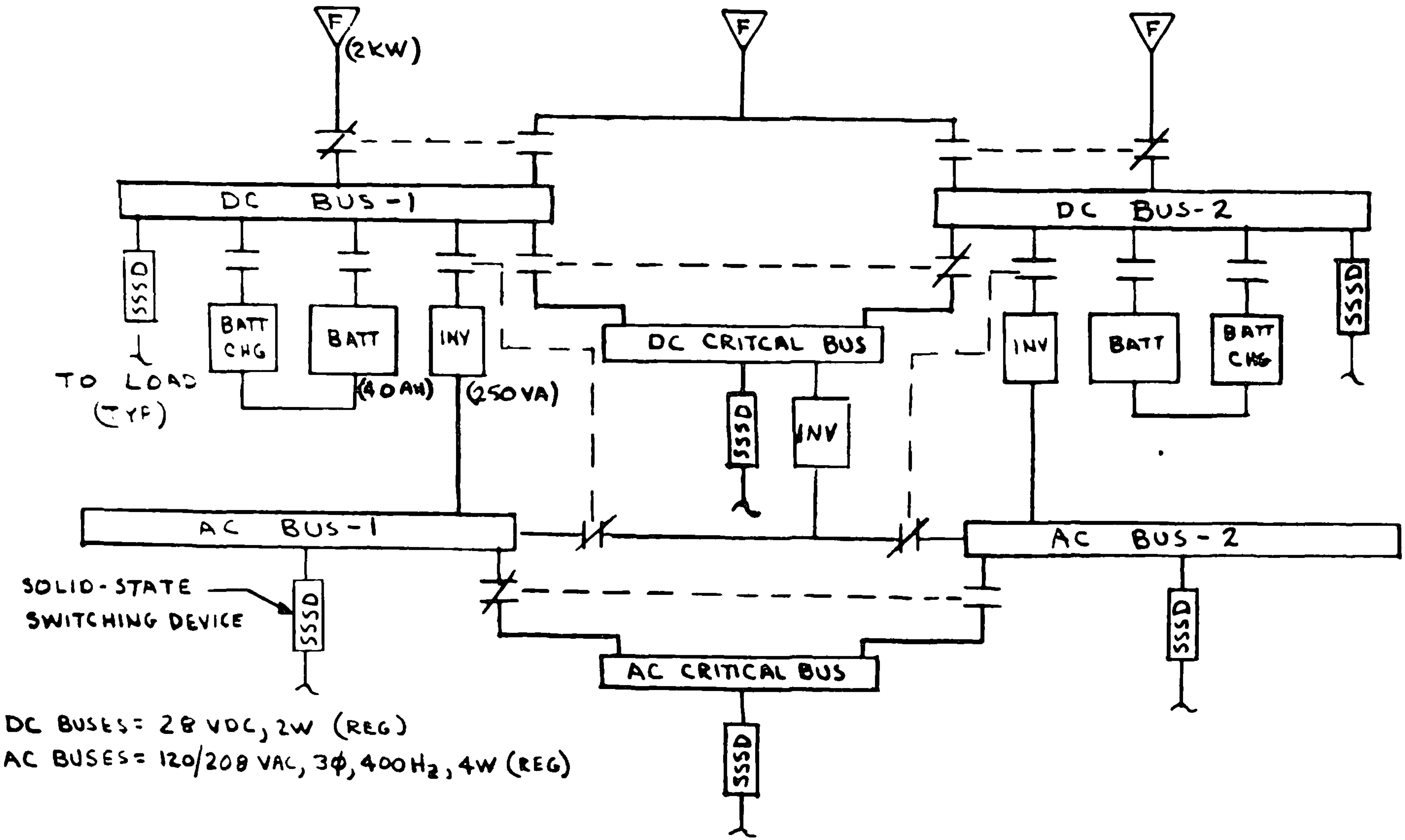
* SS = steady-state run

** PTO = NERVA run start-up through pump tailoff peak power delivery requirements to the engine, thereby reducing the weight of electrical power cables between the power source and the engine. With this configuration the peak power requirement during engine steady-state run would be approximately 700 watts. Battery recharge power requirements are anticipated to be 360 watts for a period of ten hours following a major engine burn (ANSC's ALM mission). The cooldown power requirement would remain at 200 watts.

Electrical Power Subsystem Analysis

Three fuel cells will provide the power level, voltage regulation and redundancy capability for supplying the primary direct current (dc) power needs of the RNS. Two of the three fuel cells would be normally connected to the busses with the third fuel cell in a ready standby mode. This configuration will allow increased subsystem life capability by alternate utilization of the three fuel cells as well as providing redundancy capability. Three, three-phase, 400 Hertz inverters are included for altering current (ac) loads in the otherwise dc power subsystem configuration. Two silver-zinc (one-year lifetime) secondary batteries will supplement the fuel cells for peak loads and emergency power requirements. Power is delivered to the ac and dc control centers for distribution throughout the vehicle with due regard for electromagnetic compatibility considerations. Solid-state switching devices and circuit breakers will be used for switching and control where practical. Cooling of solid-state conversion equipment and batteries will be effected through mounting on cold plates. Radiators, mounted externally on the structure of the astrionics bay are provided for fuel cell cooling. Where practical, the electrical power components will be located in a common equipment bay.

Ratings of the major components are shown in the Figure 5-17 schematic. The fuel cells are rated at 2 kw each and are based upon an average RNS load as high as 1.9 kw, with transient peak loads at engine start increasing to 5.75 kw. The total power level required during steady-state engine run will be slightly less than 4.4 kw (includes 10% allowance for losses). Since operating times at the higher power levels and transient peak power levels are relatively short compared with the longer lower average power level, two fuel cells will be capable of supplying power for all other RNS mission operations.



DC BUSES = 28 VDC, 2W (REG)
AC BUSES = 120/208 VAC, 3ϕ, 400 Hz, 4W (REG)

Figure 5-17. EPS Single Line Schematic

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SAFETY/ORDNANCE

This subsystem provides a positive means of terminating vehicle flight in the event of excessive deviation from the planned trajectory. Since the RNS is passive (payload to the INT-21) throughout the boost phase, the signal activating the ordnance for propellant dispersion of the RNS is sent from the S-IC or S-II with the initiators located on these vehicles. Therefore, passivation of this system is not required after separation from S-II in earth orbit.

Therefore, the entire subsystem consists maintly of two linear shaped charges, for redundancy, running the length of the tank sidewalls. The ordnance is located within the Systems Tunnel. Controlled detonators fuses (CDF) interface with the S-II through quick disconnects. Figure 5-18 presents a schematic of the subsystem.

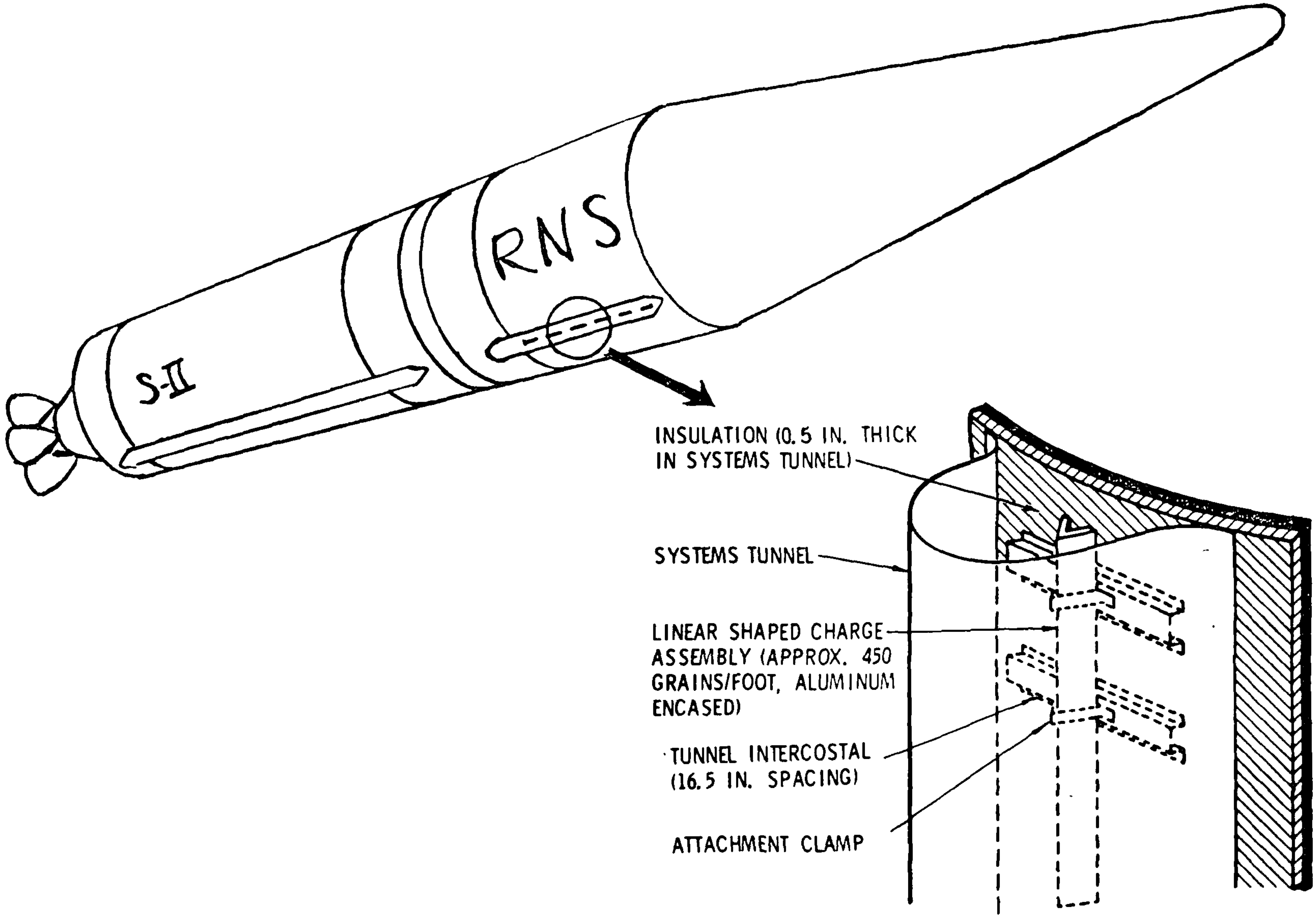

Figure 5-18. Tank Propellant Dispersion Assembly

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5.2 Pressurization Gas Requirements, ANSC Report, SO54, 201, SO54-CP090290-Fl, November 1970.

5.3 Compressed Gas Handbook, NASA SP-3045, 1969, U.S. Government Printing Office, Washington, D.C. 20402.

5.4 Miller, J. S., et al, Final Report Multiprocessor Computer System Study, Intermetrics, Inc., Contract NAS9-9763 (March 1970).

5.5 Stark, Charles, "A Fault-Tolerant Information Processing System for Advanced Control Guidance, and Navigation," R-659, MIT (May 1970).

6.0 MASS CHARACTERISTICS

This section summarizes the estimated and calculated weights for the single tank conical aft bulkhead configuration. Computer programs were employed as applicable to synthesize the weight of each component. Loads, environment, operational life, mission, ground and space operations requirements as well as the candidate materials, fabrication, installation, duty cycle, and geometric characteristics formed the basis for the analysis. In addition, existing hardware data were modified as required. Moreover, in all the calculations and assumptions, consideration was given to the penalties associated with the installation of each component. This is essential to assure realistic weight for performance and cost studies. Items included were joints, splices, supports, cutouts, interconnects, umbilicals, harnesses, fittings, valving, quick disconnects, redundant hardware, and component attachments. Where "hard data" were not available, a weight factor based on experience was assigned to maintain realism in the results. The magnitude of each factor was determined taking into consideration not only design criteria and installation, but also shape, location, and function of the component as well. For example, skirts and thrust structure fabricated of the same material exhibit completely different installation factors established by the loading, shape, and location. That is, the thrust structure shape and necessary cutouts and equipment supports carry the highest installation weight penalty of all the structural components. The forward and aft skirts follow in severity. The variation in installation weight between these two components also depends on stage design configuration, thrust structure arrangement, and equipment location.

The astrionic subsystem weights including electronics and electrical power components are based on existing hardware and/or manufacturer's data, as applicable.

Table 6-1 presents the weight summary for the baseline vehicle following the format established by NASA-MSFC. Table 6-2 presents a detailed weight breakdown for the Reaction Control System. Also presented in the table are the volume, pressure, and conditioning requirements for the oxygen and hydrogen reactants.

Table 6-3 presents a detail weight breakdown of the astrionic system including electrical power, environmental control, and propellant manage-

Table 6-1

SINGLE TANK DESIGN RNS WEIGHT SUMMARY
(CONICAL AFT BULKHEAD)

CODE	DESCRIPTION	- ORBITAL ASSEMBLY - INVERTED TANK LAUNCH	
		Lunar Mission	Ground Launch
2.00	Structure	(24325)	(33325)
2.01	Propellant Tank	17470	17470
	Fwd Bulkhead	1710	1710
	Cyl Section	6900	6900
	Aft Bulkhead	8860	8860
2.02	Engine Thrust Structure	500	500
2.03	Forward Skirt	2080	2080
2.04	Aft Skirt	1100	1100
2.05	Tunnel & Fairings	1025	1025
2.06	Exterior Finish & Sealer	430	430
2.07	Equipment Support Structure	780	780
2.08	Astrionics Module Structure	940	940
2.09	Additional Structure	N/A	(9000)
	Aero nose cone (expendable)		9000
3.00	Meteoroid/Thermal Protection	(13120)	(13120)
3.01	Insulation	7220	7220
3.02	Meteoroid Protection	5900	5900
4.00	Docking/Clustering	(1320)	(1040)
4.01	Fwd Docking System	520	520
4.02	Aft Docking System	800	520
4.03	Clustering Structure	N/A	N/A
5.00	Main Propulsion	(30225)	(2950)
5.01	NERVA Engine *	27230	N/A
5.02	External Disc Shield for NERVA**		N/A
5.03	Purge System & Leak Detection	280	280
5.04	Propellant Scavenging Sys & Sensors	N/A	N/A
5.05	Propellant Feed System	560	320
5.06	Pressurization System	1150	1130

* Reference NERVA Engine Reference Data - Sept. 1970.
** 4050-pound external shield required for manned missions.

Table 6-1 (contd)

CODE	DESCRIPTION	- ORBITAL ASSEMBLY - INVERTED TANK LAUNCH	
		Lunar Mission	Ground Launch
5.07	Fill & Drain/Orbit Refueling	655	655
5.08	Ground & Emergency Vent	N/A	N/A
5.09	Flight Vent	350	565
6.00	Auxiliary Propulsion	(1320)	(1320)
6.01	Reaction Control System	1320	1320
6.02	Retro System	N/A	N/A
6.03	Ullage System	N/A	N/A
7.00	Astrionics System/Astrionics	(5055)	(5055)
7.01	Guidance Navigation & Control **	870	870
7.02	Instrumentation	445	445
7.03	Command & Control	N/A	N/A
7.04	Electrical Power ***	2060	2060
7.05	Electrical Network	350	350
7.06	Environmental Control	575	575
7.07	Propellant Management	600	600
7.08	On-board Checkout	45	45
7.09	Data Management	110	110
8.00	Safety Ordnance System	(175)	(175)
8.01	Safety System	95	95
8.02	Ordnance System	80	80
9.00	Contingency	0	0
	SUBTOTAL	75540	56985

** Includes a 500 pound allowance for the NERVA instrumentation and control subsystem.

*** This weight includes fuel cell reactants

Reactant Breakdown:	Usable	=	1030 lb
	Residual	=	50 lb
	Total		1080 lb

Table 6-1 (Continued)

CODE	DESCRIPTION	- ORBITAL ASSEMBLY - INVERTED TANK LAUNCH	
		Lunar Mission	Ground Launch
10.00	RCS Propellant	(5800)	(5800)
11.00	Residual Propellant	(2890)	(1100)
11.01	Liquid Propellant	0	0
11.02	Vapor Vented	(included in Code 13.00)	
11.03	Vapor	2890	1100
12.00	Reserve Flight Performance	(included in Code 14.00)	
13.00	Propellant Boiloff	(1360)	0
14.00	Impulse Propellant (Startup, Main-stage, Shutdown, Cooldown, Reserve)	(295750)	N/A
14.01	First Burn		
14.02	Second Burn		
14.03	Third Burn		
14.04	Fourth Burn		
15.00	Usable LH_2 Placed in Orbit	N/A	186165
	TOTAL	381340	250000

Lunar Mission B.O. Weight

Subtotal	75540	
Less usable fuel cell react.	1030	
Plus Residual Vapor	2890	
B.O. Weight	77400	lb at end of last cooldown (unmanned missions)
Plus External Radiation Shield	4050	
B.O. Weight	81450	lb at end of last cooldown (manned missions)

NERVA Engine Orbital Assembly 27470 lb

Table 6-2. Supercritical Storage RCS Requirements

	O_2	H_2
Propellant Weight Lbs	4600	1200
Storage Tank Weight Lbs	280	730
Accumulator Weight Lbs	37	92
Heat Exchanger Weight Lbs	14	18
Gas Generator Weight Lbs	5	7
Line and Valve Weight Lbs	44	33
Subtotal Lbs	4980	2070
Engine Weight Lbs	70	
Total System Weight Lbs	7190	
Storage Tank Pressure	800	500
Accumulator Pressures	800/450	500/200
Storage Tank Volume Ft^3	68	285
Accumulator Volume Ft^3	8.2	33.6
Conditioning Temperature ^oR	300	200

ment subsystems. The propellant management in this case consists of the sensing system required for propellant gauging under a positive g. It is of importance to note that the weight distribution shown reflects operational, 1974 state-of-the-art equipment. All necessary electronic components, hardware, and cabling are included. Installation and support structure, except as noted, is reported as structure under Code 2.08 in Table 6-1, RNS Weight Summary. Guidance, Navigation and Control subsystem details are as defined in the subsystem design section discussed previously. Other sources employed in the detailed definition of the installed components include S-II and Apollo historical data, as well as the Space Station Study, and available manufacturer's data.

The Guidance, Navigation and Control system includes a 500-pound NERVA digital instrumentation and control subsystem (NDIC) as specified in the NERVA Engine Reference Data, September 1970.

Telemetry and measuring estimates are based on vendor data and S-II type components projected to 1974 state of the art. This projection assumes a volume reduction of 33 percent and a weight reduction of 50 percent due to microminiaturization. The system is multiplexed and uses S-band system antennas.

Table 6-3. Astrionics System Detail Weight Statement
(Single Tank Baseline Design)

Code	Description	Weight (Lb)		
7.00	Astrionics System/Astrionics			(5055.0)
7.01	Guidance, Navigation & Control			870.0
	Inertial Measurement Unit (IMU)		60.0	
	IMU - Strapdown System	58.0		
	Spare Power Supply	2.0		
	Star Trackers (3)		64.5	
	Tracker Heads (2)	24.0		
	Electronics (2)	18.0		
	Optics	22.0		
	Coaxial Cabling	0.5		
	Horizon Sensor		57.5	
	Tracker Head Electronics	49.0		
	Optics	8.0		
	Coaxial Cabling	0.5		
	X-Band Radar		20.5	
	Transmitters (2)	4.5		
	Receiver	5.0		
	Antenna	7.0		
	Antenna Extension System	3.0		
	Coaxial Cabling	1.0		
	Scanning Laser Radar (SLR)		55.5	
	Target			
	Sensor	20.0		
	Electronics	6.0		
	Chaser			
	Sensor	20.0		
	Electronics	9.0		
	Coaxial Cabling	0.5		
	Stationkeeping Radar		33.0	
	Transmitters (2)	4.5		
	Receivers (4)	16.0		
	Omni-directional Antennas (4)	0.5		
	Coaxial Cabling	12.0		
	Radar Transponder		10.0	

Table 6-3. Astrionics System Detail Weight Statement
(Single Tank Baseline Design)
contd.

Code	Description	Weight (Lb)		
	Television		11.0	
	Camera	10.0		
	Coaxial Cabling	1.0		
	NDIC (including container)		500.0	
	Data Busses (coaxial)		58.0	
	Within A.U. (2)	25.0		
	NDIC to Engine (2)	33.0		
7.02	Instrumentation			445.0
	Communications		71.0	
	S-Band	(50.0)		
	Transmitter (2)	8.0		
	Receiver	5.0		
	Omni Antennas (4)	0.5		
	Directional Antenna & Drive Mechanism	19.5		
	Antenna Extension	5.0		
	Coaxial Cabling	12.0		
	X-Band	(5.5)		
	Electronics	5.0		
	Coaxial Cabling	0.5		
	UHF	(15.5)		
	Transmitters (2)	1.0		
	Receiver	2.0		
	Antennas (4)	0.5		
	Coaxial Cabling	12.0		
	Telemetry & Measuring		374.0	
	Electronics, Multiplexer, Modulators (2)	101.0		
	Transducers	55.0		
	Signal Conditioners, Amplifier, Power Division	185.0		
	Cabling	33.0		
7.03	Command and Control		--	
7.04	Electrical Power			2060.0
	Fuel Cells (3)		360.0	
	Batteries, Chargers & Boxes		70.0	
	Filters, Heaters, Compressors		50.0	

Table 6-3. Astrionics System Detail Weight Statement
(Single Tank Baseline Design)
contd.

Code	Description	Weight (Lb)		
	Radiator		50.0	
	Plumbing		40.0	
	Valves, etc.		40.0	
	Inverters (3)		40.0	
	Tankage Including Supports		330.0	
	GOX (1)	130.0		
	GH_2	200.0		
	Reactants		1080.0	
	GOX	960.0		
	GH_2	120.0		
7.05	Electrical Network			350.0
	Junction Boxes		50.0	
	Conditioning Equipment		50.0	
	Cabling		250.0	
	Within Astrionics Unit	150.0		
	To Engine (2 Leads)	100.0		
7.06	Environmental Control			575.0
	Passive Sys Assumed Weight Allocation			
	Accts for Containers (except for NDIC), Insul, and Heaters			
7.07	Propellant Management			600.0
	Electronics		25.0	
	Sensors		185.0	
	Mast		250.0	
	Cabling & Supports		140.0	
7.08	On-Board Checkout			45.0
	Remote Acquisition & Control Units (9)		45.0	
7.09	Data Management			110.0
	Computer		100.0	
	I.O. (2)		10.0	

The electrical power system weights represent the fuel cells plus batteries arrangement discussed previously in this volume. The electrical network is based on S-II data. Cabling within the astrionics bay is assumed to equal approximately that within the S-II forward skirt. Cabling for furnishing power to the NERVA is estimated at 33 percent of the S-II electrical cabling unit weight in the tunnel.

Propellant management weights are based on S-II fuel gauging data and consist of point sensors, capacitance probes, sensor mast, electronics, cabling, and support structure. The S-II system weighs approximately 315 pounds. For the RNS weight allocation, sensors and mast were scaled as a function of tank length, and cabling and supports assumed to be 30 percent of the total. Zero-g gauging is not included in the breakdown pending further definition.

The weight breakdown shown, therefore, represents an optimistic design approach taking into consideration the knowledge of the operational requirements of each component.

Table 6-4 presents the stage mass and inertia characteristics employed to establish the Reaction Control System requirements.

Table 6-4. RNS Mass and Inertia Properties

Propellant Loading	Mass (Slugs)	Roll Inertia Slug $ft^2 \times 10^6$	Pitch and Yaw Inertia Slug $ft^2 \times 10^6$	Center of Gravity (feet)
NO PAYLOAD				
Empty	2,720	0.36	4.2	70.0
Full	12,100	0.36	12.0	51.5
1/2 Full	7,400	0.36	8.0	61.0
PAYLOAD ATTACHED				
Full	15,800	0.67	26.0	31.5
*Reference datum plane at station 1869 (Reference Figure 5-3)				

7.0 STAGE INTERFACES

Some of the more significant flight interfaces that the RNS stage will potentially encounter include the NERVA, Space Shuttle, RNS payload, Propellant Depot, Maintenance Element, and Space Tug. These will be treated in this section with the exception of the payload and Propellant Depot. Lack of definition of these two potential systems preclude a valid assessment at this time. The major pre-operational interface that the stage encounters is with the INT-21, discussed in Section 8 of this volume.

As previously mentioned, the stage less engine is boosted to orbit in an inverted position employing the INT-21 booster. Mating and assembly with the Space Shuttle delivered NERVA is performed in low earth orbit. Interface with a Maintenance Element on a scheduled basis is performed prior to each mission to accomplish flight readiness tasks including checkout, as well as components repair and replacement. This interface also can take place at various points during the mission depending on unscheduled maintenance requirements of the vehicle. The space tug could interface with the stage for payload delivery as well as for engine disposal as discussed in Volume II - Part A - Sections 2 and 3 of this report.

NERVA/STAGE INTERFACE

Both physical and functional interface requirements have been identified for the NERVA/stage and are discussed below.

Physical Interface

Physical interface between the NERVA and stage encompass the areas of structures, fluid/mechanical interconnects, and electrical/electronic interconnects for ground and flight test as well as for the operational program.

The recommended test program will be accomplished with flight design hardware. Consequently, the physical interface requirements for the test and operational programs will be the same with the exception of cooldown and purge requirements peculiar to the ground test program only. These exceptions have not been identified at present.

Both standard and modified engine designs were employed to ascertain a satisfactory interface of the 8-degree half cone-angle, 25-inch cap radius tank with the NERVA. For example, Figure 7-1 design illustrates the feasibility of incorporating a standard neuter docking system in the engine/tank interface for automatic orbital mating of the engine and tank. The latest engine configuration is used, with the following modifications: the main feed lines are joggled inboard to accommodate the docking system geometry and to facilitate connection to the 25-inch radius tank cap; and six large electrical connectors of the same total pin area are employed in place of the 46 smaller connectors as specified in the latest ANSC reports. The active docking assembly is incorporated in the stage thrust structure, and a passive docking ring is attached to the engine forward thrust plate. An actuated coupling plate inside of the active docking section on the stage provides for holdback of the line and electrical connections during structural acquisition and lockup, with subsequent positive alignment and coupling of the subsystems.

If the turns in the main propellant lines at the TPA inlets are objectionable, the TPA's can be rotated to align with the propellant lines without significant effect on the overall configuration. Joggling of the lines has the advantage of bringing them closer to the engine gimbal center, thus, reducing the amount of flexibility required to accommodate the engine gimbal motions. If access for inspection, etc., of the forward side of the interface is desired, the thrust structure can be constructed as an open truss rather than a monocoque shell.

In order to maximally exploit the radiation shielding advantage of the small (25-inch) aft cap radius of the baseline tank, it would be desirable to contract the engine accessory systems ahead of the external shield - particularly the TPA's - which may be significant secondary scatterers. In the current engine configuration, the TPA's are located 35 inches out from the centerline and the valves/plumbing project beyond the nominal 50-inch external shield radius in many places. The results of preliminary studies presented in Figures 7-2, and 7-3 have indicated that, with a nominal lengthening of the thrust structure ahead of the shield, the TPA's can be pulled in to 22 inches from center and the valves/plumbing contracted to a 38.5-inch radius envelope without otherwise significantly altering the present arrangement and component concepts. The depicted interface design employs this contracted engine configuration.

The stage thrust structure, docking, and coupling provisions are the same as for the standard engine configuration with joggled feed lines. The principal difference is that the feed lines do not require joggling. If the engine design can, in fact, be so modified, additional shield weight savings might be accrued and a simpler interface design accomplished.

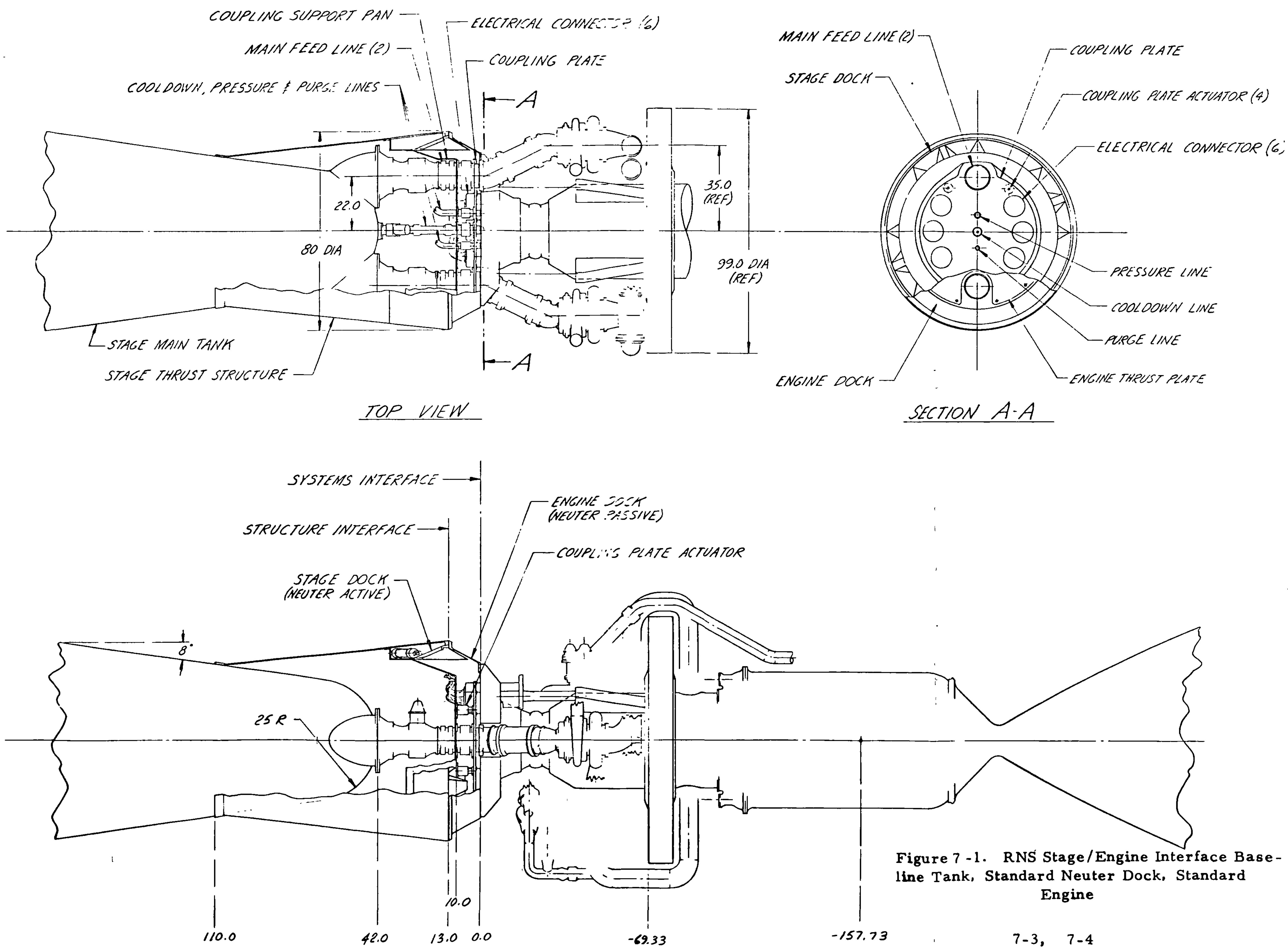


Figure 7-1. RNS Stage/Engine Interface Baseline Tank, Standard Neuter Dock, Standard Engine

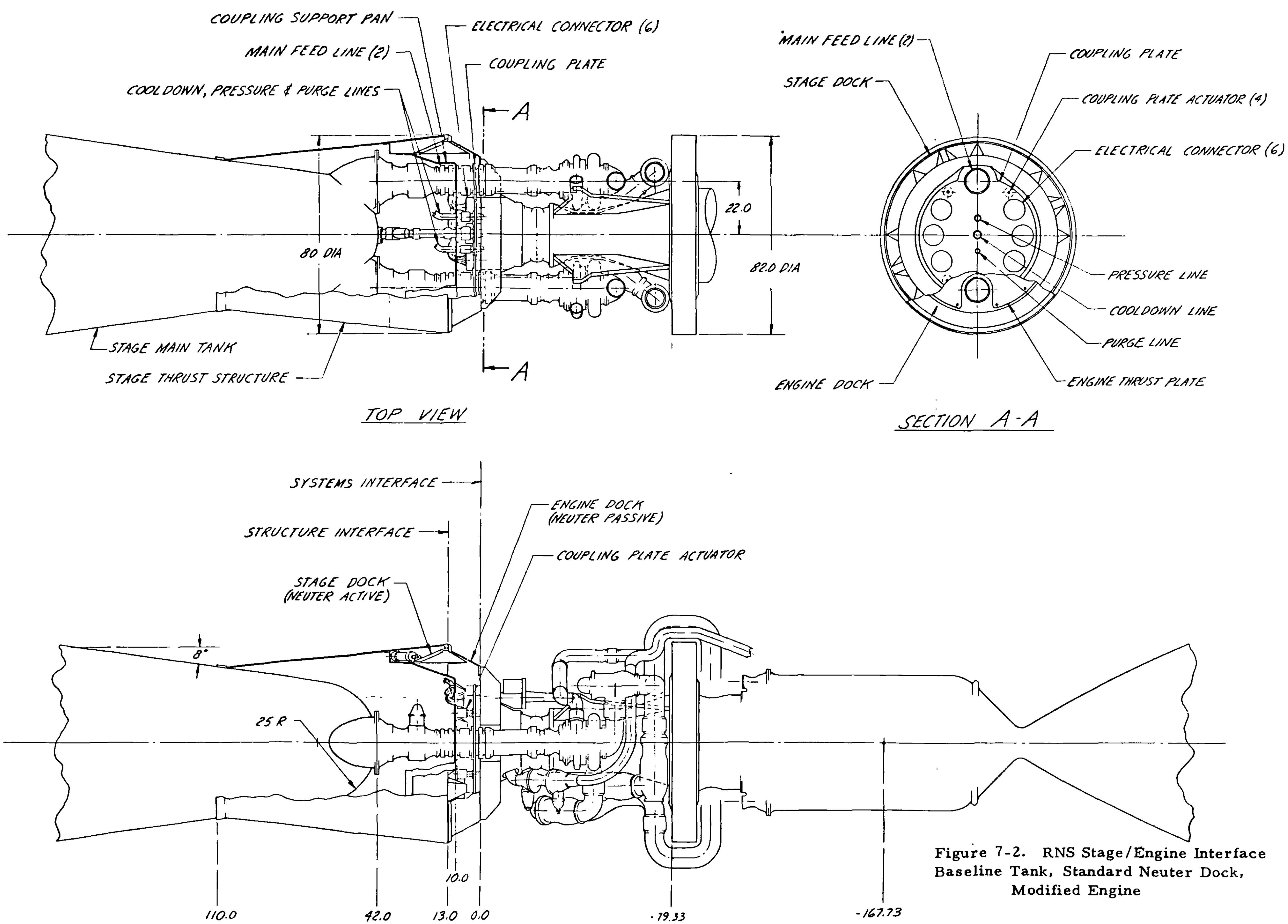


Figure 7-2. RNS Stage/Engine Interface Baseline Tank, Standard Neuter Dock, Modified Engine

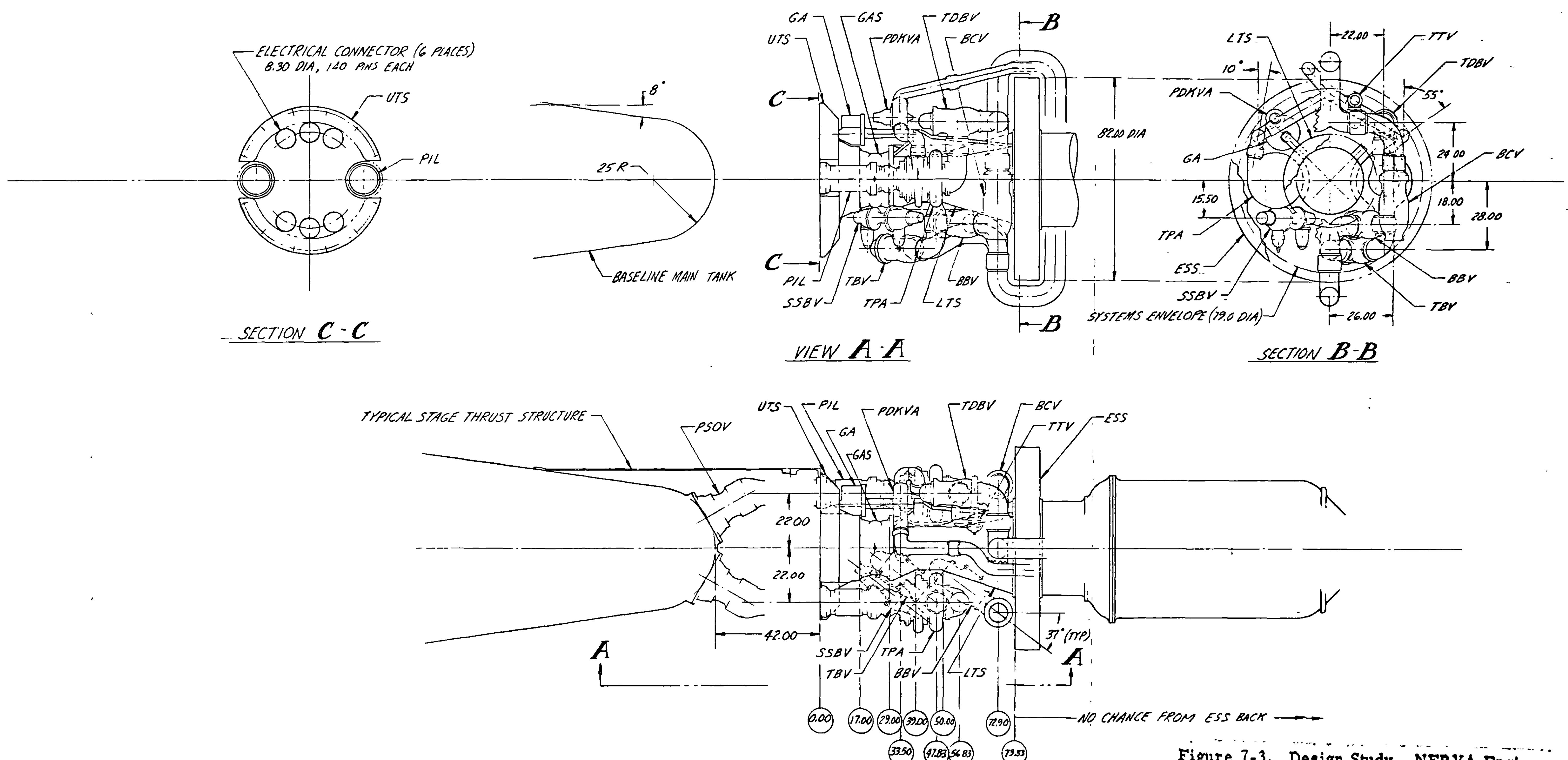


Figure 7-3. Design Study - NERVA Engine Modified Accessory Envelope

Structural

The structural interface between the stage and the NERVA consists of the two segments of the Neuter Dock (described in Section 5.0 of this volume under Docking) i.e., an active ring and cone assembly mounted on the stage thrust structure and a passive ring anchored to the forward thrust structure of the NERVA as shown in Figure 7-4. This basic concept was devised by NR/Space Station Program to accommodate other IPP elements and provide where applicable a sealed passage way between docked modules.

Fluid/Mechanical Interconnects

The fluid interconnects shown in Figure 7-1 include propellant lines at the PSOV's, cooldown line to the cooldown-valve module, engine purging, and stage tank pressurization. In addition, temperature conditioning for the engine I&C electronics may be required.

Propellant is fed to the NERVA engine via two propellant feed lines 9.5 inches diameter each through the propellant shutoff valves. Cooldown propellant is supplied through the 3.0-inch diameter cooldown line and is controlled by the cooldown flow valve. The autogenous pressurization line recommended by NR is 2.25 inches in diameter.

Electrical/Electronics Interconnects

The stage electrical interfaces defined by ANSC in E105-CP090-290-F1, "Engineering Data for Design Evaluation," September 1970, per Engine concept 1137400C, consist of a total of 46 panel-mounted electrical connectors at the stage interface based on the current measurements-requirements list and incorporation of the I&C signal-conditioning multiplexer units in the engine. Details of the interface are given in Figure 7-5. The specific connector design has not been completed. Provisions will be included for remote mating of the electrical connectors. At the engine/stage interface (station zero) 480 power contacts and 633 total contacts on connectors or total conductors have been identified.

As presented earlier in Figures 7-1, -2, and -3, NR/SD proposes to reduce the number of connectors at this interface to six, down from the 46 identified by ANSC, retaining, however, the same total pin area. This approach is suggested to facilitate the alignment of the connectors and simplify the interface.

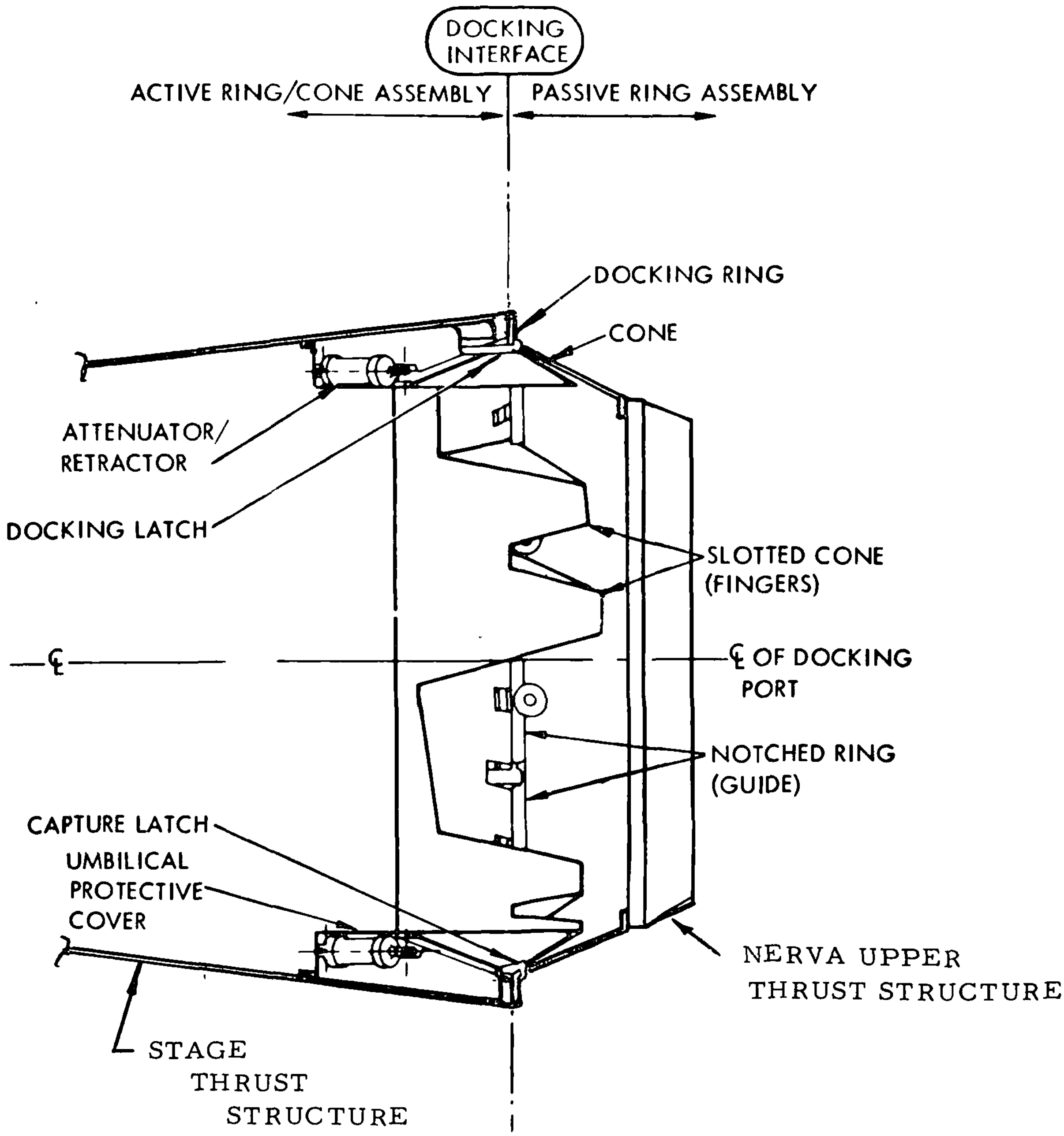
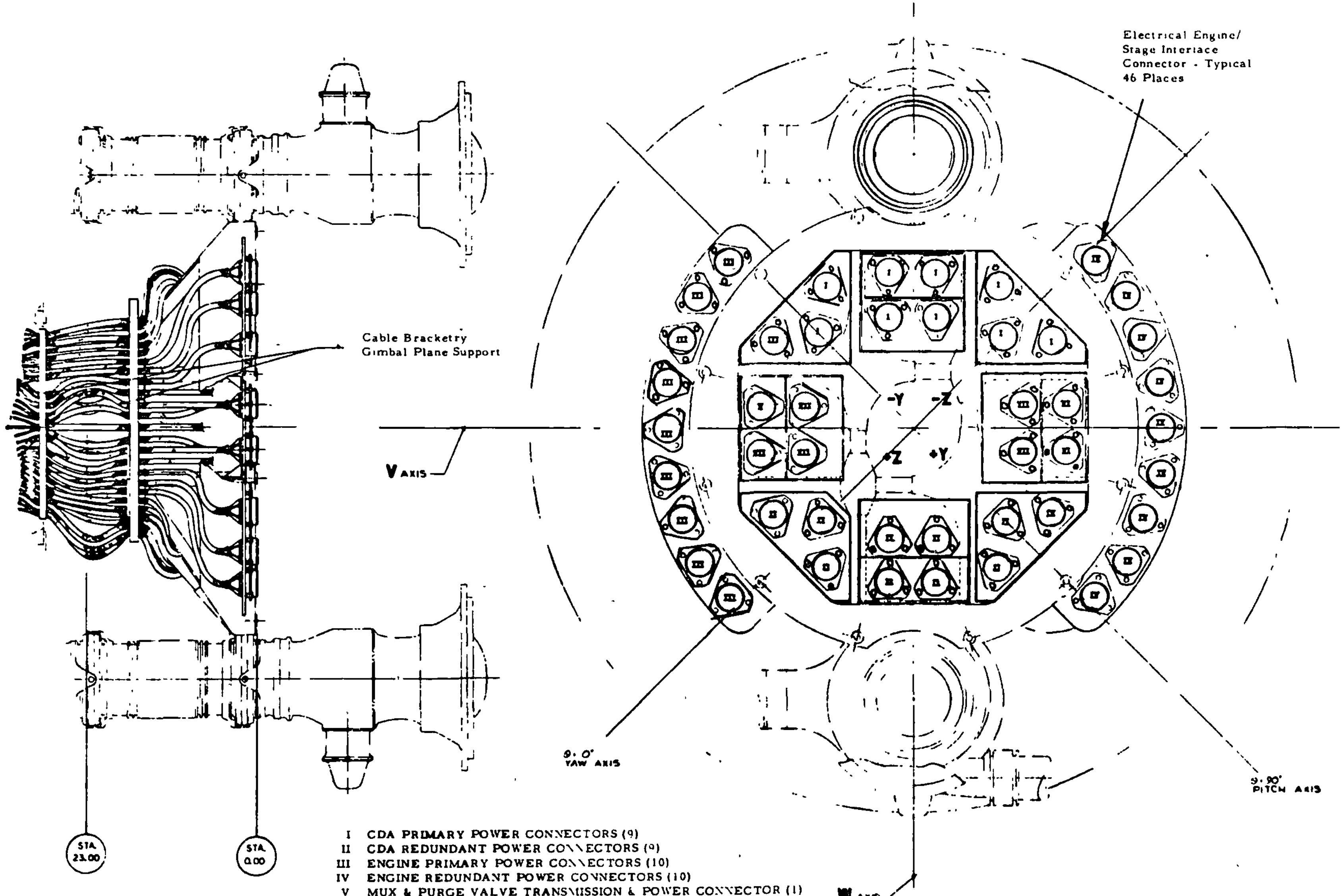


Figure 7-4. Docked Configuration, Standard Docking



- I CDA PRIMARY POWER CONNECTORS (9)
- II CDA REDUNDANT POWER CONNECTORS (9)
- III ENGINE PRIMARY POWER CONNECTORS (10)
- IV ENGINE REDUNDANT POWER CONNECTORS (10)
- V MUX & PURGE VALVE TRANSMISSION & POWER CONNECTOR (1)
- VI MUX & PURGE VALVE REDUNDANT TRANSMISSION & POWER CONNECTORS (2)
- VII NEUTRONICS & SPARE CONNECTORS ([illegible])

Figure 7-5. Electrical/Electronics Interconnects

Functional Interface

The functional interface between the engine and stage include thrust and inertial loads transmitted through the neuter docking system/ thrust structure assemblies, as well as environmental conditioning of the engine compartment during ground test operations. Also included are: fluid requirements, operating pressure and temperature, flow rates, and electrical/electronics requirements.

Loads

The interface loads include both engine run, and docking loads.

Engine Run - The limit loads associated with this phase of engine operation are:

Axial = 75000 pounds (NERVA NOMINAL THRUST)
Shear = $\pm$ 5600 pounds (based on resultant deflected angle of 4.26° in the corner of a square pattern resulting from 3° actuator rotation)
Moment = $\pm$ 210,000 in-lb (static only, dynamic TBD)

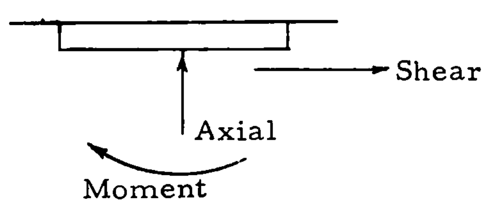
Docking Loads - The neuter dock described in the Physical Interface Section is employed in the docking operations between the engine and stage. The system has the following capabilities:

Axial Miss Distance: $\pm$ 5.0 inches
Angular Misalignment: $\pm 4^{\circ}$
Rotational Misalignment: $\pm 4^{\circ}$
Closing Velocity at Impact: 0.5 FPS

The loads are as follows:

Axial = TBD
Shear = TBD
Moment = TBD

Sign Convention



Engine Purge and Environmental Conditioning

Vehicle Engine Compartment - Some degree of environmental conditioning of the engine when confined within the Space Shuttle cargo bay compartment and during cold flow and hot firing tests appears to be a necessity. This could include isolation of the engine from environmental loading during the launch and boost phase and may also include temperature and atmospheric conditioning during this time as well as during tests. Further definition of these requirements will be supplied as information becomes available.

Engine System Purge/Conditioning - The engine contains an integral purge unit which consists of a system of tubing to distribute purge gas from an external supply to components requiring purge. The engine is capable of venting and purging as required to prevent the accumulation of explosive concentrations of hydrogen and to provide environmental protection to sensitive engine components, again, during ground test operations. It is possible to purge the engine by an external source of inert gas during this phase.

A continuous purge of inert gas may be required for storage periods: (1) storage in a controlled environment, 5 years; (2) launch pad environment, 6 months. Purging will not be required in space. There is also no requirement to purge the engine before and after firing. Flow rates and quantities of gas required are not known at this time. This information will be supplied when available.

Engine system purge fluids are as follows: (1) GN_2 procured to MSFC Specification 234; and (2) GHe procured to MSFC Specification 364.

Instrumentation and Control Subsystem Purge/Conditioning - All electronic packages will require purging, during ground testing and prelaunch, to prevent entrapment of a mixture of hydrogen and air. The only units to be exempt from this requirement are those which will be hermetically sealed. Flow rates and pressures have not been established. Temperature conditioning is required during flight operations.

The units to be considered for purge and/or conditioning are: (1) NERVA digital I&C electronics; (2) valve actuator drivers (locations to be determined later); and (3) multiplexers.

NERVA digital I&C electronics will require purging with gaseous nitrogen during ground test and pre-launch operations to prevent a hydrogen and air mixture. When the system is operating during flight, it will require conditioning to $70^{\circ}F \pm 30^{\circ}F$.

Conditioning for Engine Operation

The NERVA engine is designed to operate with liquid hydrogen which meets requirements of MSFC Specification 356A. Engine performance is based upon propellant delivered to the turbopumps at the conditions outlined in the pump performance map in Figure 7-6.

Performance of the NERVA turbopumps is shown in the figure as a plot of Net Positive Suction Pressure and % Vapor versus propellant flow rate and engine thrust. The data show that hydrogen can be pumped under a variety of conditions (including two phases) over a range of NPSP's. The nominal design condition is a flow rate of 46 lb/sec per pump for hydrogen at 28 psia saturation pressure, zero NPSP. The design goal is 26 psia saturation pressure, zero NPSP. As an example of the flexibility available, the figure shows that hydrogen at 20 psia saturation pressure can be pumped at 46 lb/sec per pump provided an NPSP of 2.9 is available (that is, tank outlet pressure is 22.9 psia).

These data were utilized in determining tank pressure for hydro/thermal management. Analysis of the various burns showed that for the Passive Propellant Management recommended by NR, the critical design condition is malfunction operation (single pump operation) for the last burn (EOI). At this time, nuclear and solar heating has heated the LH_2 to 23.5 psia saturation pressure (39.5 R). The 80-percent thrust requirement for malfunction can be met by 3.5 NPSP, that is, by a tank outlet pressure of 27 psia. However, at some degradation of specific impulse, malfunction operations can be achieved at 65-percent of rated thrust. ANSC is considering this thrust level for possible throttle mode operations. The figure shows that single pump operation at 65-percent thrust (59.5 lb/sec) requires only 1.5 NPSP for propellant at 23.5 psia saturation pressure (e.g., a pressure of 25 psia at the tank outlet). It has been determined that 2.5 psia is required to account for flow losses into the outlet line, flow losses through the capillary devices in the tank, and vent valve setting band width. Thus, required tank design is 27.5 psia.

In addition to pump performance during steady burn, pump operation interfaces with stage design during preconditioning and bootstrapping. At the lower pump rotation rates during start-up, vapor ingestion is not as critical; however, present ANSC analysis shows that the lowest hydrogen temperature acceptable for bootstrapping is 36.8 R (15 psia saturation pressure).

Figure 7-7 presents the ullage pressure and liquid temperature history for the reference lunar mission.

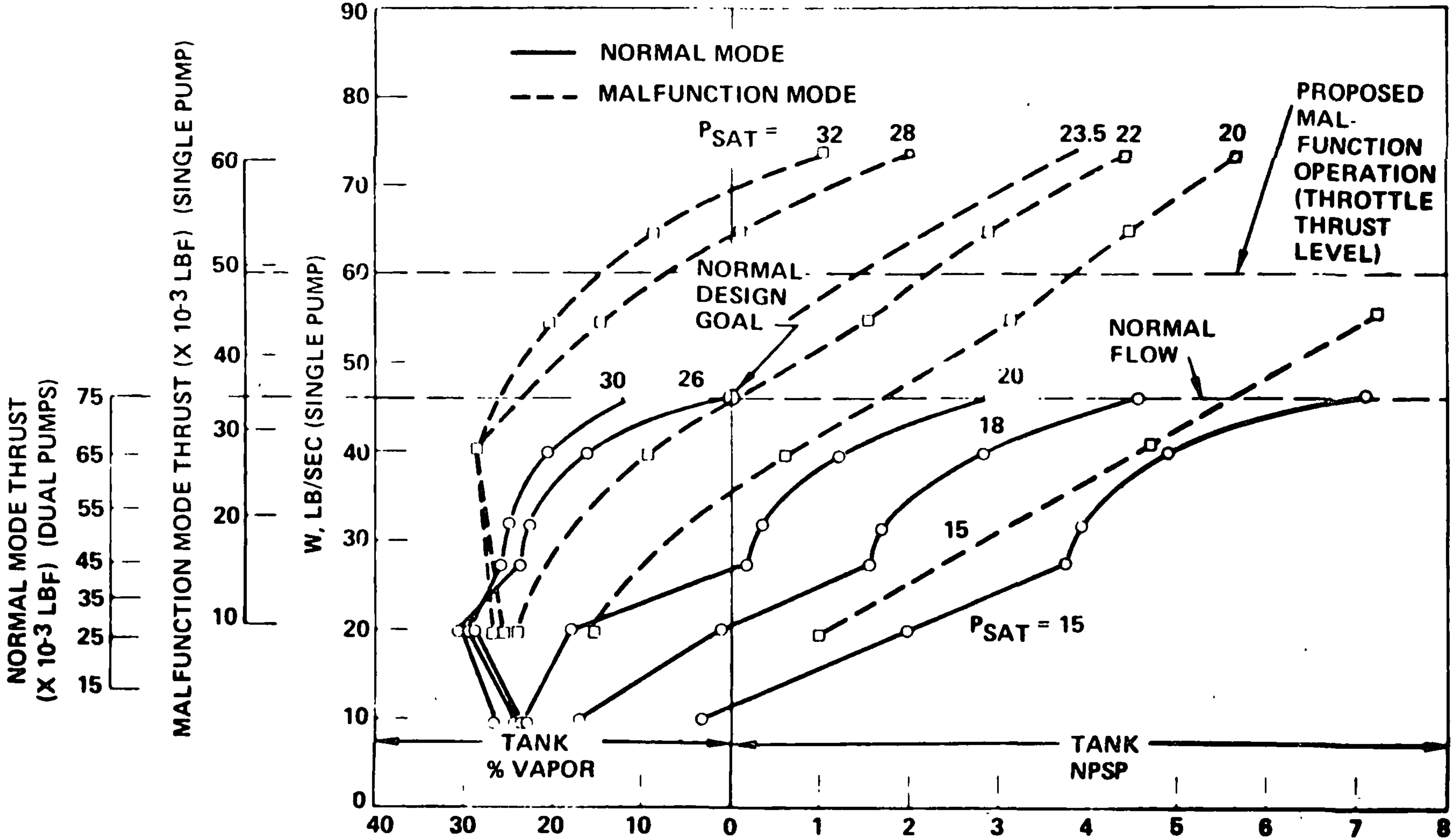


Figure 7-6 Pump Performance Map

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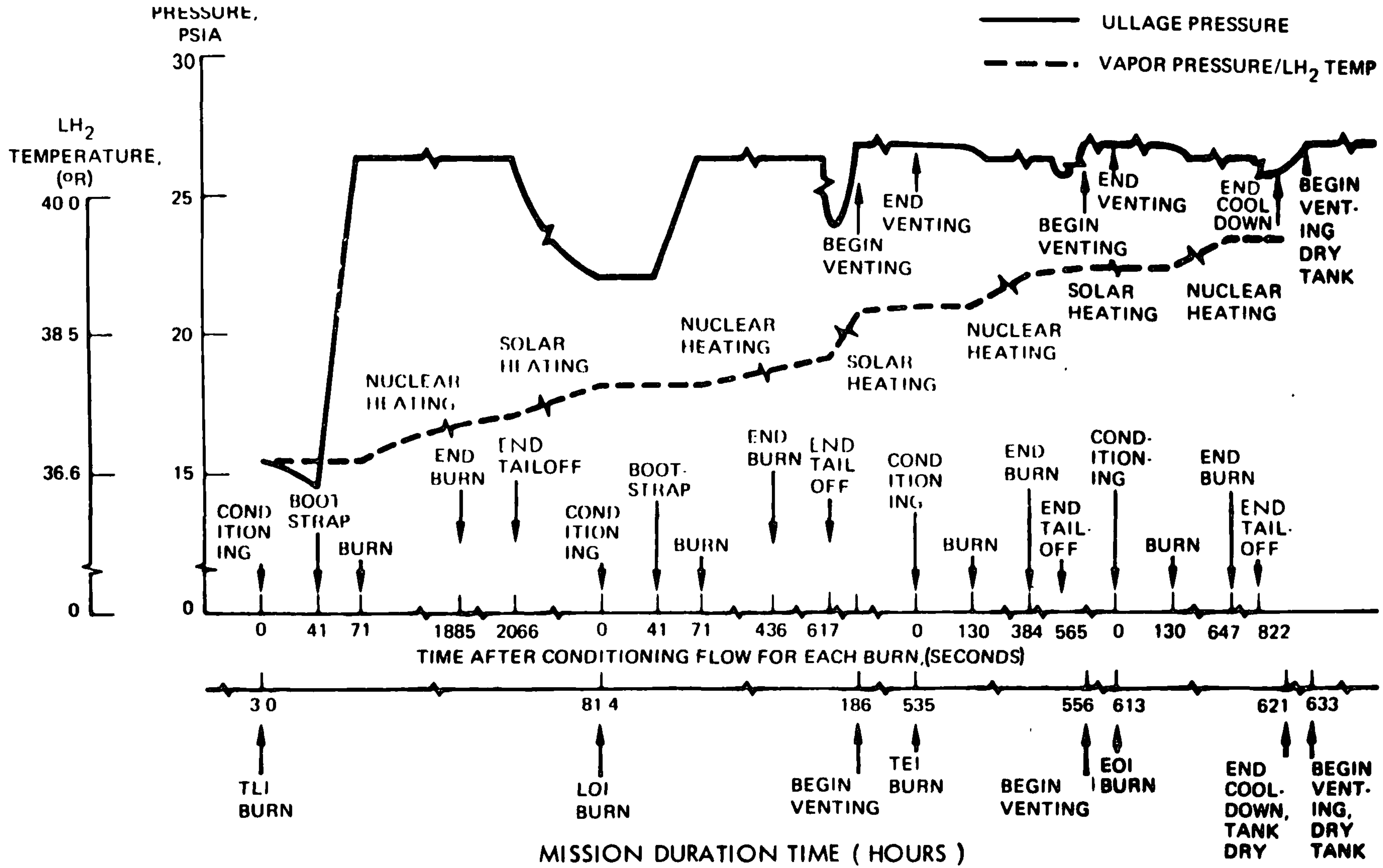

Figure 7-7 Ullage Pressure & Liquid Vapor Pressure Profiles

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Analysis of thermodynamic behavior during engine chilldown prior to the TLI burn has been based upon a total flow of 280 lbs of hydrogen in 40 seconds. Latest information from ANSC indicates that the chilldown time is now approximately 130 seconds, but that the total flow of 280 lbs is a reasonable value.

The analysis was conducted using the following initial conditions: ullage pressure = 15.5 psia, liquid temperature = 36.8 °R (saturated), ullage volume = 5% or 3400 cu. ft. The results are shown in Figure 7-8. This shows that as chilldown flow occurs, ullage pressure drops. The concomitant vaporization that can occur at the pump inlet is also shown on the figure. By the end of the conditioning period, just prior to bootstrapping ullage pressure has dropped by 0.8 psia to 14.7 psia, with a potential for vaporization at the pump inlet of 11%.

The pump performance curves previously shown reveal that 11% vapor formation is tolerable at the low RPM's during pump start-up. However, requirements for the ANSC pump allows only zero percent vapor at start-up. There are a number of ways around this difficulty. They include (1) use of heat leak to raise tank pressure, with use of tank heat exchanger to maintain liquid at 36.8 °R, (2) use of a propellant depot to prepressurize the tank prior to the first burn, and (3) updating pump requirements for propellant quality to anticipated levels.

Another potential problem is that of the drop in propellant temperature to the equilibrium value at 14.7 psia (Figure 7-7). ANSC analyses show that the minimum acceptable temperature is the equilibrium value of 15 psia. This problem can be surmounted in a variety of ways including requiring that the propellant temperature after refueling correspond to the equilibrium value at 16 psia. Whatever fix is used, this potential problem must be considered in establishing engine/stage interface requirements in future studies.

Conditioning During Cooldown Phase

Figure 7-9 illustrates propellant flow rates requirements during cooldown. This figure shows cooldown pulse profiles following translunar injection (TLI) for a single burn and a two-burn TLI. Continuous flow at either 1.7 or 0.4 lb/sec is required immediately after engine run and for some time thereafter (e.g., about 2,000 seconds following TLI); subsequently and for the greater part of the cooldown on-off pulsed flow is required at 0.7 lb/sec. The no-flow portion of the on-off cooldown phase increases with increasing time, so that during the bulk of the cooldown phase, a zero "g" condition exists.

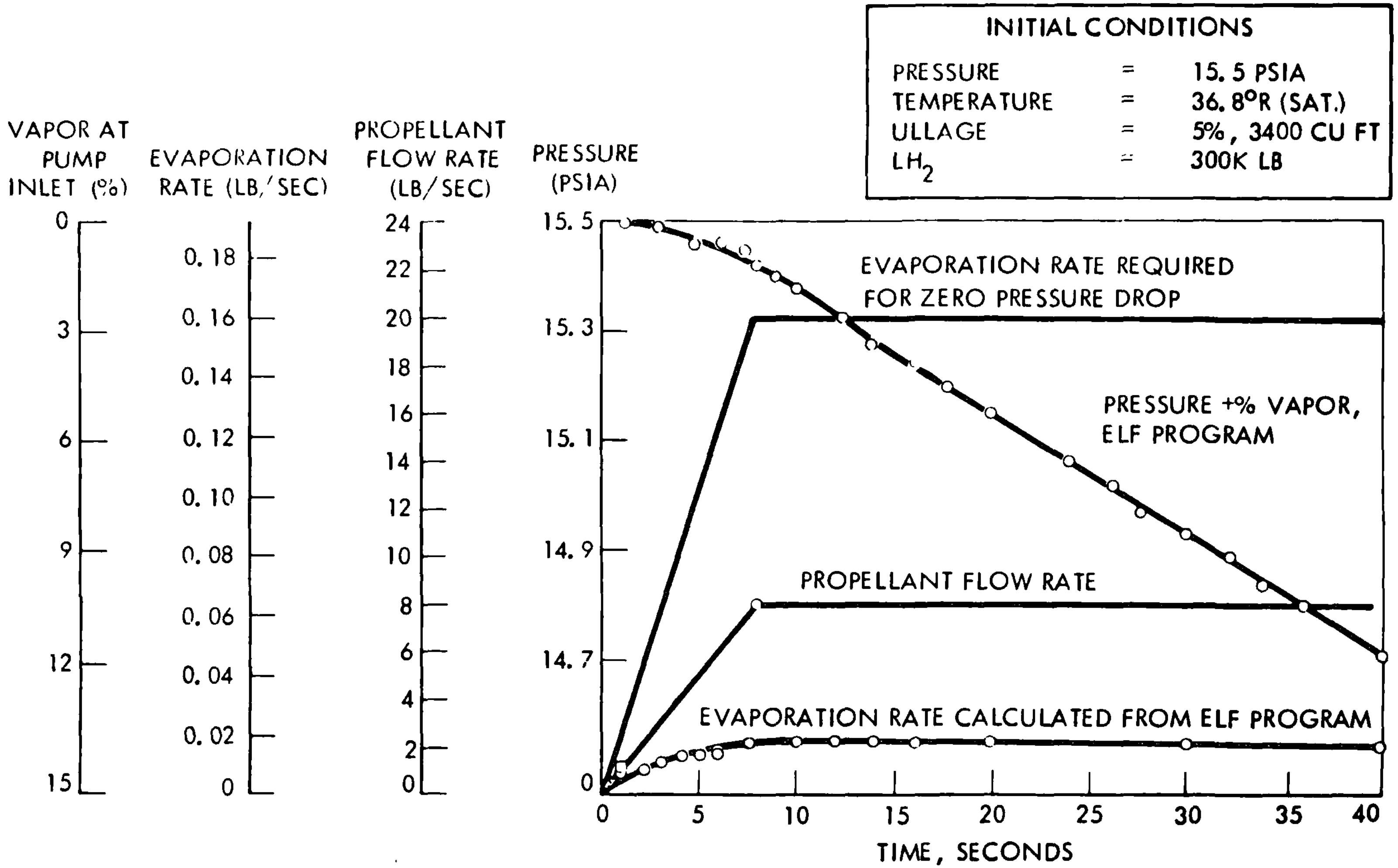

Figure 7- 8 Pressure Drop and Evaporation Rate during NERVA Conditioning - First (TLI) Burn

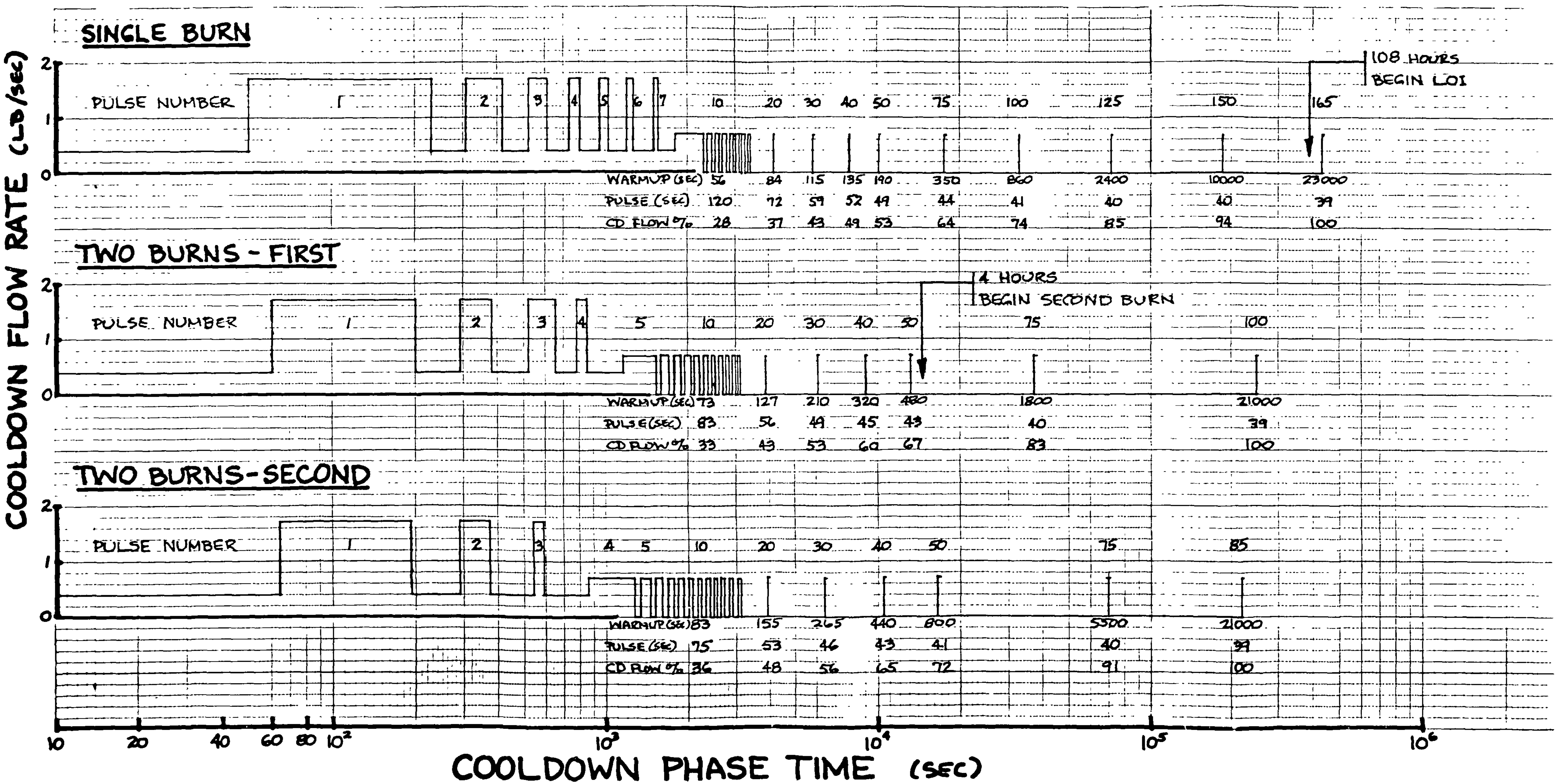

Figure 7-9 Cooldown Pulse Profiles Following Translunar Injection

The approach taken to provide propellant feedout at zero quality and proper pressure during cooldown is to utilize capillary channels and rewetting screens in conjunction with the passive propellant management system designed for the vehicle.

The cooldown compartment in the tank supplies the engine cooldown propellant during the mission. This compartment is sized for "worst case" hydrodynamics for the lunar mission. Cooldown following the TLI burn presents no problems as the bulk propellant compartment is essentially full. Therefore, liquid flow to the cooldown compartment and, henceforth, to the cooldown line is assured. Likewise, gas free liquid flow occurs at LOI burn initiation. The situation can be different after the LOI burn, as the bulk propellant compartment is then partially depleted. This compartment will be settled by the early cooldown flow (0.4 and 1.7 lb/sec). However, during later cooldown flow (0 and 0.7 lb/sec pulsed flow), propellant could be dislocated to the upper end of the bulk propellant compartment. This would lead to vapor passage from the bulk compartment to the cooldown compartment, vapor passage to the restart compartment is precluded by the wicking capability of the V-shaped wicking clusters. These clusters are sized to provide cooldown flow to the restart compartment under zero to minus 10^{-5} g's.

Bulk propellant can still be dislocated in the bulk and restart compartments at TEI burn initiation. The wicking clusters cannot provide adequate flow at engine initiation and, therefore, the cooldown compartment must be settled by RCS propellant prior to the burn. It has been determined that 40 pounds of RCS propellant is sufficient to settle the restart compartment. As this is not sufficient to settle the bulk propellant, a small amount of vapor could enter the cooldown compartment while thrust build-up occurs. During TEI steady burn, it is expected that vapor will be "burped" out of the cooldown compartment at the barrier vertex. This is the basis for the small cone angle and large open area specified for the cooldown barriers.

This same sequence of events will occur during TEI cooldown and EOI restart. A determination of the maximum amount of vapor that could accumulate in the cooldown compartment has been made. No allowance was made for mitigation of this problem by vapor burping during steady burn. A total of 5,350 lbs of liquid could be displaced by vapor due to adverse location during cooldown flow, withdrawal of thermal conditioning fluid from the cooldown line, and restart flow. Therefore, the cooldown plus restart compartments have been sized for a LH_2 capacity of 5,350 pounds plus 5,950 pounds for a total of 11,300 pounds. The 5,950 pounds LH_2 requirement is the maximum propellant needed for the TEI cooldown.

The lowest operating pressure during cooldown is not expected to decrease below 21.8 psia. This occurs at the end of the TLI cooldown.

Pressurization

Pressurization is a key engine/stage interface. Autogenous pressurization is employed in the RNS. This system utilizes engine heat to vaporize some of the LH_2 pumped from the tank; this hydrogen gas is then returned to the tank, maintaining the tank's pressure as the expulsion continues. A turbopump system is used to feed propellant to the engine with turbine exhaust working fluid, supplied via a by-pass from the engine, being used as pressurant. The start-up phase when engine chamber pressure, pump spin rate, and turbine exhaust supply are building up is the critical operational period. Pressurant supply rate requirements are appreciably higher during this period than during steady engine burn, rate requirements surging to stay ahead of pump requirements.

In performing the pressurization analysis before ANSC's start-up analysis had been completed it was necessary to make some assumptions as to tank pressure requirements. In making this first cut, it was assumed that tank pressure rise during bootstrapping was linear with time.

Figure 7-10 shows pressurant flow rate requirements as a function of time for a single TLI burn. Pressurant flow is initiated at bootstrap pressurization and reaches a peak value of 1.15 lb/sec just prior to achievement of steady burn pressure. Pressurant requirements during steady burn range from 0.66 to 0.61 lb/sec during the 1690 seconds steady burn period - about half the peak value. Pressurant flow is continued during throttled burn and tail-off to maintain pressure during this period.

Proceeding to the LOI burn, a comparison of the perfectly mixed ullage with stratified ullage as it affects pressurant flow rate has been made. For the former case, it was found that after mixing the propellant and ullage gas (after the TLI burn) the equilibrium pressure was only slightly above 15.5 psia; this is due to the large amount of evaporation required to supply the ullage with saturated vapor.

Pressurant flow rate is shown in Figure 7-11 for three cases. Curve A is a repeat of pressurant flow rate for TLI burn (Figure 7-10) and is presented for comparison purposes. Curve B shows the pressurant flow rate for the LOI burn for the case of the ullage remaining stratified after the TLI burn. Here, as previously discussed, the ullage pressure had fallen to 22.8 psia just prior to the LOI burn. Curve C shows pressurant flow rates for an LOI burn where tank contents had mixed sufficiently to attain thermodynamic equilibrium. Initial tank pressure in this case is 15.5 psia.

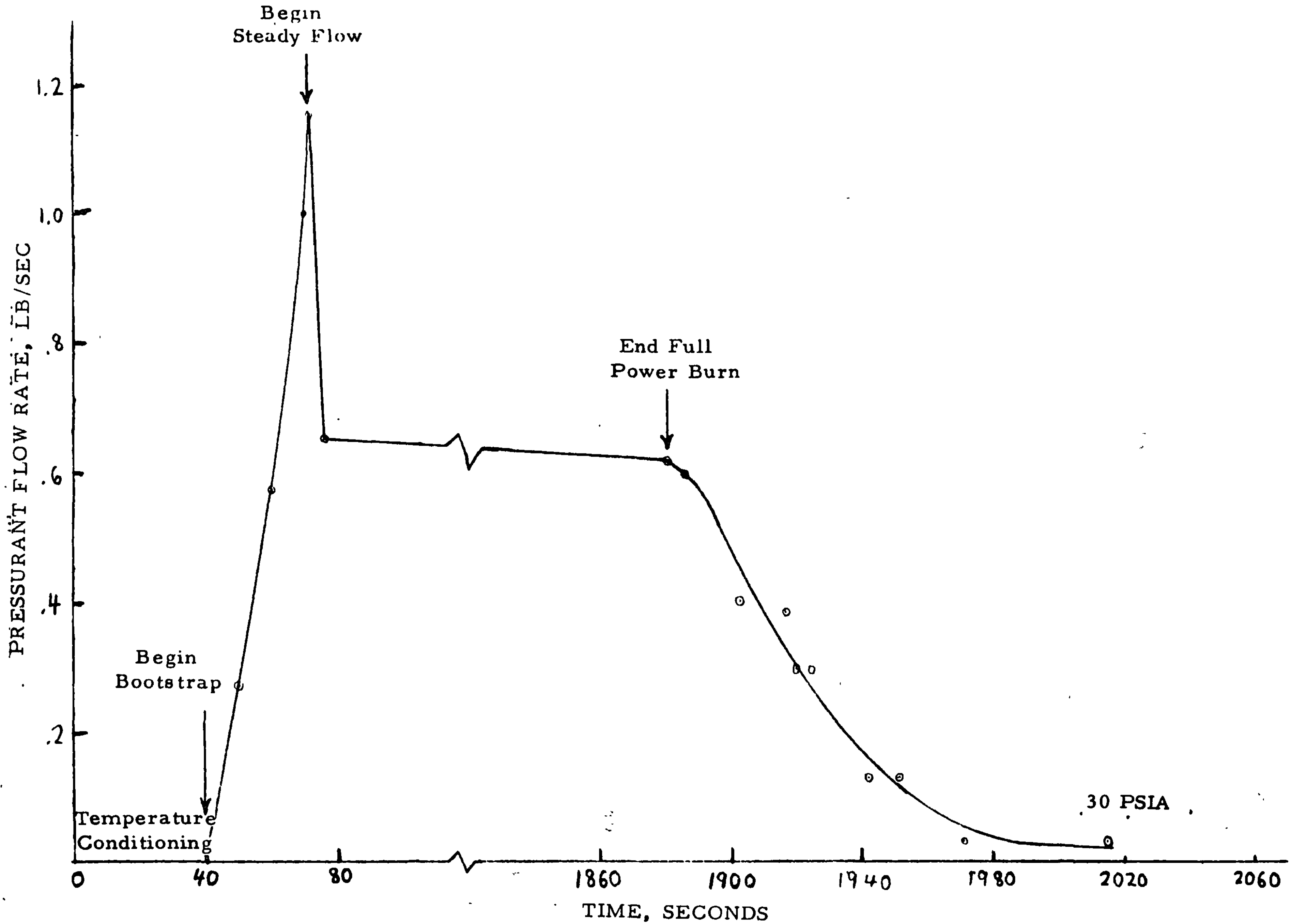

Figure 7-10 Pressurant Flow Rate Profile
Single Burn TLI

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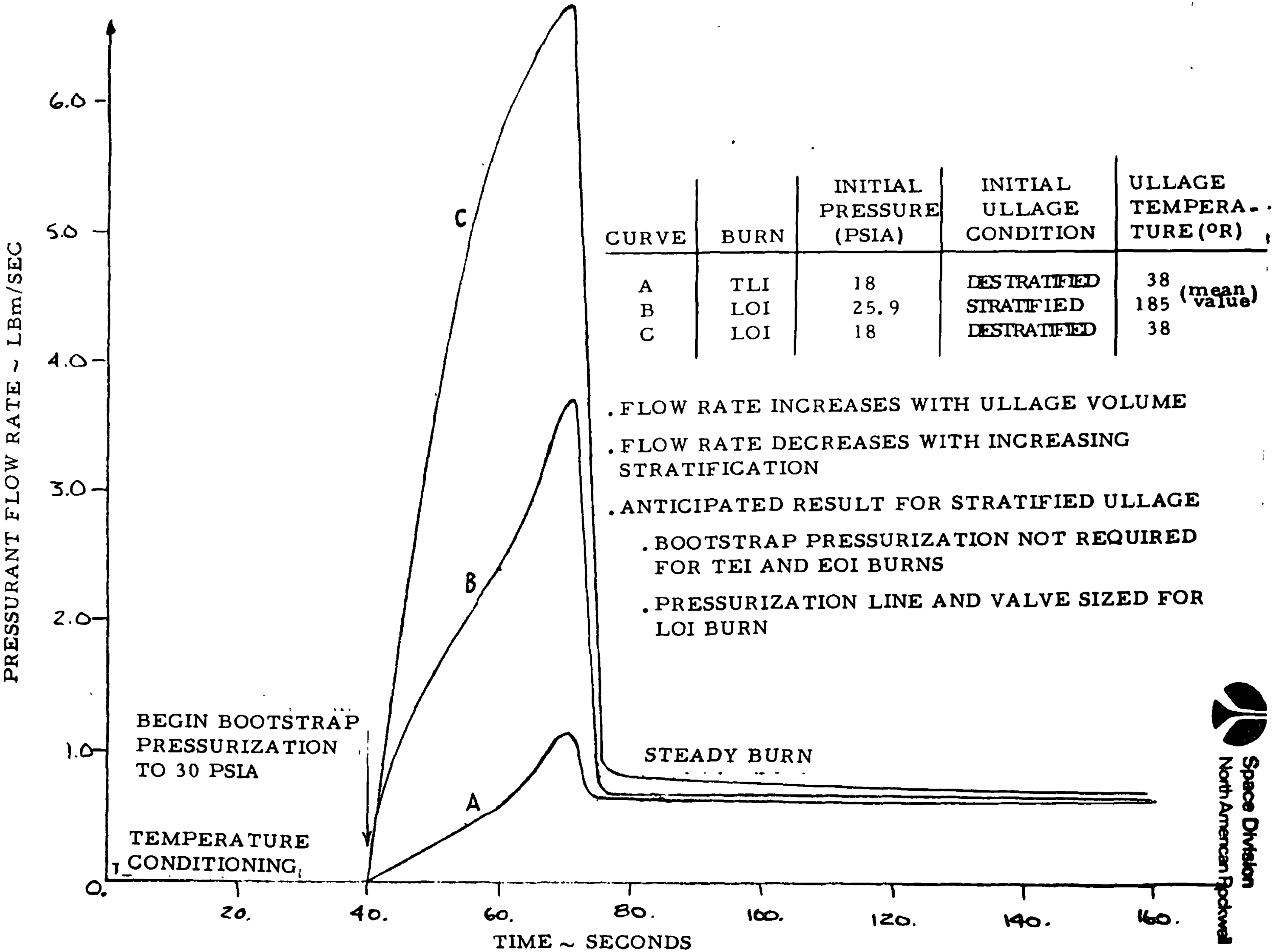

Figure 7-11 Variation in Pressurant Flow Rate for TLI & LOI Burns

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Peak pressurant flow rate is highest for Curve C (about 6.8 lb/sec) because of the large initial ullage and low initial pressure. Even though the initial pressure is high (22.8 psia) for the stratified LOI burn, peak pressurant flow rate is 3.4 lb/sec which is three times as high for this case than for the TLI burn; this is due to the much larger initial ullage for the LOI burn.

Bootstrap pressurization is not required for the TEI and EOI burns for the passive (stratified) propellant management. As passive propellant management is the recommended method, Curve B of Figure 7-11 is the appropriate one to use in line sizing.

To account for uncertainties as to degree of ullage stratification, flow rate upon which the design was based was increased from 3.4 to 4.5 lb/sec. Equations and design charts for compressible pipe flow with friction were used in the analysis.

Conditions at the turbine exhaust are, pressure = 650 psia, temperature = 260°F. Using the previously mentioned design chart, it was determined that a line size of 1.5 inch would be adequate. However, telephone conversations with ANSC personnel revealed that the assumption of a linear pressure rise is not appropriate. Their detailed study of bootstrap dynamics has shown that the pressure range is more nearly exponential with high pressurant rate requirements at the end of pressurization. Further calculations showed that a 2.25 inch line diameter would be satisfactory even for this more stringent tank pressure ramp.

NR recommends that the pressure line diameter be 2.25 inches. ANSC recommended a 3.5 inch line diameter. This difference is readily explained, as ANSC assumed a destratified ullage, while NR's analysis is based on a stratified ullage - with ullage pressure at start-up greater than the vapor pressure. Reconciliation of these two line sizes is anticipated in subsequent study phases as better analytic tools are brought to bear on ullage thermal process and more accurate assessment is made of the degree of ullage stratification.

Electrical Power

ANSC Report S-130, "NERVA Engine Reference Data," September 1970, Contract SNP-1, was utilized as the source document in identifying the NERVA power requirements. The engine firing can be divided into three distinct periods from a power consumption viewpoint; (1) engine operation including start-up, steady-state, and shutdown; (2) cooldown; and (3) coast. Table 7-1 lists the energy required for engine operation, cooldown, and coast for each burn of an eight-burn lunar mission (ANSC ALM mission). Cooldown times are based upon terminating pulse

Table 7-1 Energy Requirements for ANSC Mission ALM

Burn	Engine Operation* (w-hr)	Cooldown (w-hr)	Coast (w-hr)
1	1,050	14,280	...
2	136	1,520	...
3	105	1,540	...
4	146	11,580	...
5	132	1,520	...
6	91	1,540	...
7	105	12,200	600
8	455	19,000	...
Total	2,220	63,180	600
% of Overall	3.36	95.73	0.9

Overall energy requirement for engine operation, cooldown and coast = 66,000 w-hr

* Includes startup, steady state operation, and shutdown.

cooldown when the decay energy release data decreases to 5 kw.

The power requirements for a typical engine firing include operating engine valves and control drum actuators as well as the engine control and instrumentation power. This value, when averaged over the engine operating period, is approximately 2,200 watts. Peak power during this time may be 3,500 watts at initiation of chilldown.

The power requirements during cooldown and coast are much lower than during engine operation. Figure 7-12 shows that the power requirement during cooldown is about 200 watts with additional 400 watts pulses, which represent valve operation.

Surge currents to the engine will be significant only for the actuators. Initiation of chilldown will draw as much as 128 amp for approximately five seconds during normal operation. Other surges will be 70 amp or less for a period not exceeding five seconds.

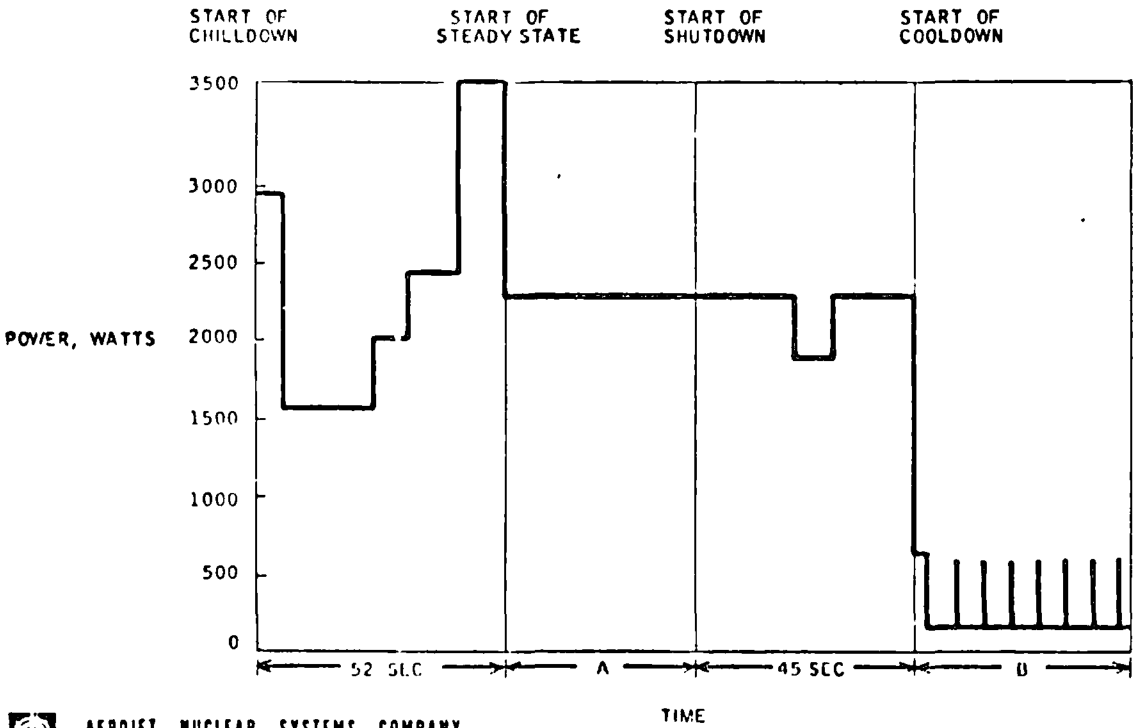
Power requirement for the engine during checkout is 200 watts and is presently identified to have a duration of 600 seconds.

The use of a battery located below the NERVA/stage interface point would reduce peak power delivery requirements to the engine, thereby reducing the weight of the electrical power cables between the power source and the engine. With this configuration, the peak power requirement during engine steady state run would be approximately 700 watts. Battery recharge power requirements are anticipated to be 360 watts for a period of ten hours following a major engine burn (ANSC's ALM mission). The cooldown power requirements would remain at 200 watts.

Telemetry

For initial flights, the Flight Measurement Requirements List (MRL) will be used to develop telemetry requirements. These requirements are divided into two major categories: (1) flight operational, including safety and trend data; and (2) flight qualification, which will be removed from the flight requirement as soon as the involved component is flight qualified.

Most flight operational data will be sampled, digitized, and processed by the engine system digital I&C electronics for the prime use of the engine system. Those digital channels designated for telemetry will be presented to the stage telemetry system as a (TBD) data train(s) at a maximum rate of (TBD) bits/sec. Each word will contain (TBD) bits for data and (TBD) bits for channel identification. Each frame will utilize one



VALUES OF A AND B FOR MISSION A-L-M		
BURN	A (MIN)	B (HR)
1	24.5	71.4
2	2.0	7.6
3	1.2	7.7
4	2.2	57.9
5	1.9	7.6
6	0.8	7.7
7	1.2	61.0
8	9.8	95.5

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Figure 7-12 - Typical Power-Demand Profile for NERVA Engine During a Single-Burn Cycle, Normal Mode word for frame synchronization.

The remaining flight data will remain in analog form and will be routed to the stage-mounted NERVA digital I&C electronics for processing, each system use, and/or transmittal to the stage for inflight use and input to telemetry. The channels designated for telemetry will be transmitted to the ground in an analog format utilizing (TBD) techniques.

It is anticipated that the stage telemetry will provide transmission of data in the following categories.

1. Digital data with sampling rates of 10 to 100 samples/sec.
2. Analog data with frequency content from 0 to 3000 Hz and total measurement system uncertainty requirements of 5 to 10%.

Radiation Environment

The applicable radiation dose levels for the various phases of engine operation are given below.

NERVA Externally Shielded - In this operating mode the engine contains a 3300 pound internal shield and a 4050 pound external shield.

Normal Operating Mode - For the normal operating mode, the radiation dose is 10 rem at tank for the reference lunar shuttle mission.

Malfunction Operating Mode - TBD

Emergency Operating Mode - TBD

After Shutdown - TBD

NERVA Externally Unshielded - The subsequent radiation levels are based on a 3300 pound NERVA internal engine shield.

Normal Operating Mode - The unperturbed iso-contours of post neutron flux densities ($E > 0.9$ Mev) and gamma kerma rates are shown in Figure 7-13 for the baseline configuration as a function of distance from the NERVA core midplane.

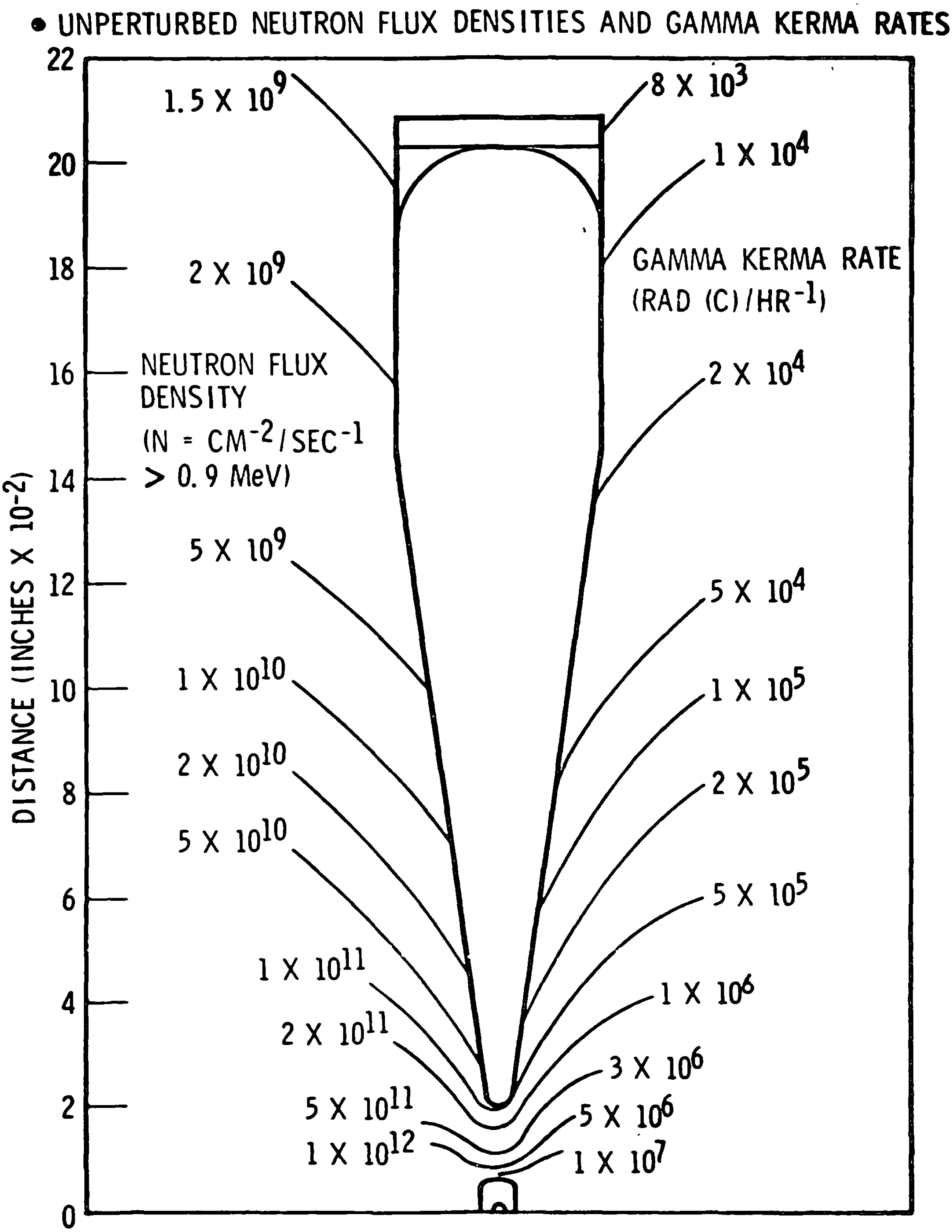

Figure 7-13 RNS Baseline Configuration - Iso-Contours

Malfunction Mode - TBD

Emergency Mode - TBD

After Shutdown - Iso-dose contours 24 hours after the NERVA firing for earth orbital insertion (EOI) phase of a representative lunar shuttle mission are presented in Figure 7-14. Gamma kerma rates (rad-hr^{-1}) are given as a function of separation distance (feet) from the NERVA core midplane with reference to the polar angle, $\alpha = 0^{\circ}$, along the vertical axis in the forward direction.

SPACE SHUTTLE INTERFACE

The NERVA is mounted in the engine delivery support structure (shaded structure shown in Figure 7-15) with the engine interface pointed forward. This orientation will take the 3g acceleration loads induced during launch. Hard points will be required near the bottom of the NERVA pressure vessel for support by the engine delivery support structure. Location of the hard point is aft of the engine center of gravity which permits handling at the c. g. during ground operations and loading. Three support points, two hard points on opposite sides of the pressure vessel and the neuter docking assembly, are provided to the engine in the shuttle.

In the concept presented in the figure, the poison wires are removed from the engine prior to deployment from the Space Shuttle by retracting the removal boom shown. This presents the problem that if mating with the stage cannot be accomplished, the engine cannot be returned to the ground facilities without creating a safety hazard. To preclude this, the engine could be supported and rotated from the nozzle end and poison wires removed only after mating with the stage is accomplished.

In the concept shown, the support structure is rotated 90 degrees and then the engine is rotated 180 degrees, pointing the passive docking ring to the stage. Following engagement of the docking system and verification of lock-up with the stage, the support structure is released and returned to the stowed position in the space shuttle cargo bay.

Following lock-up of the engine and stage, the active docking ring will be retracted, and fluid and electrical connections accomplished. This will be followed by a functional checkout employing the on-board checkout system.

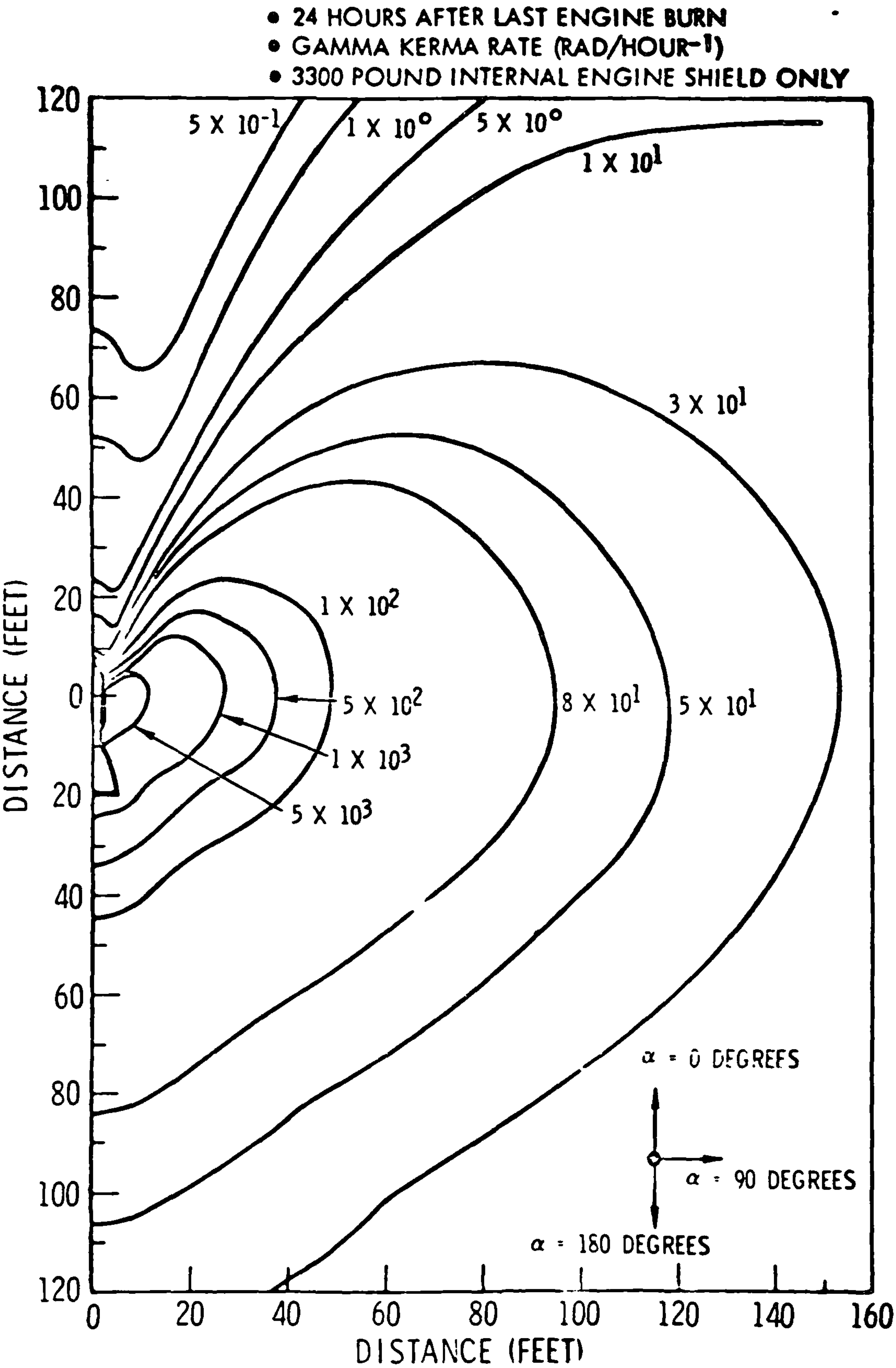

Figure 7-14 After-Shutdown Iso-Dose Contours

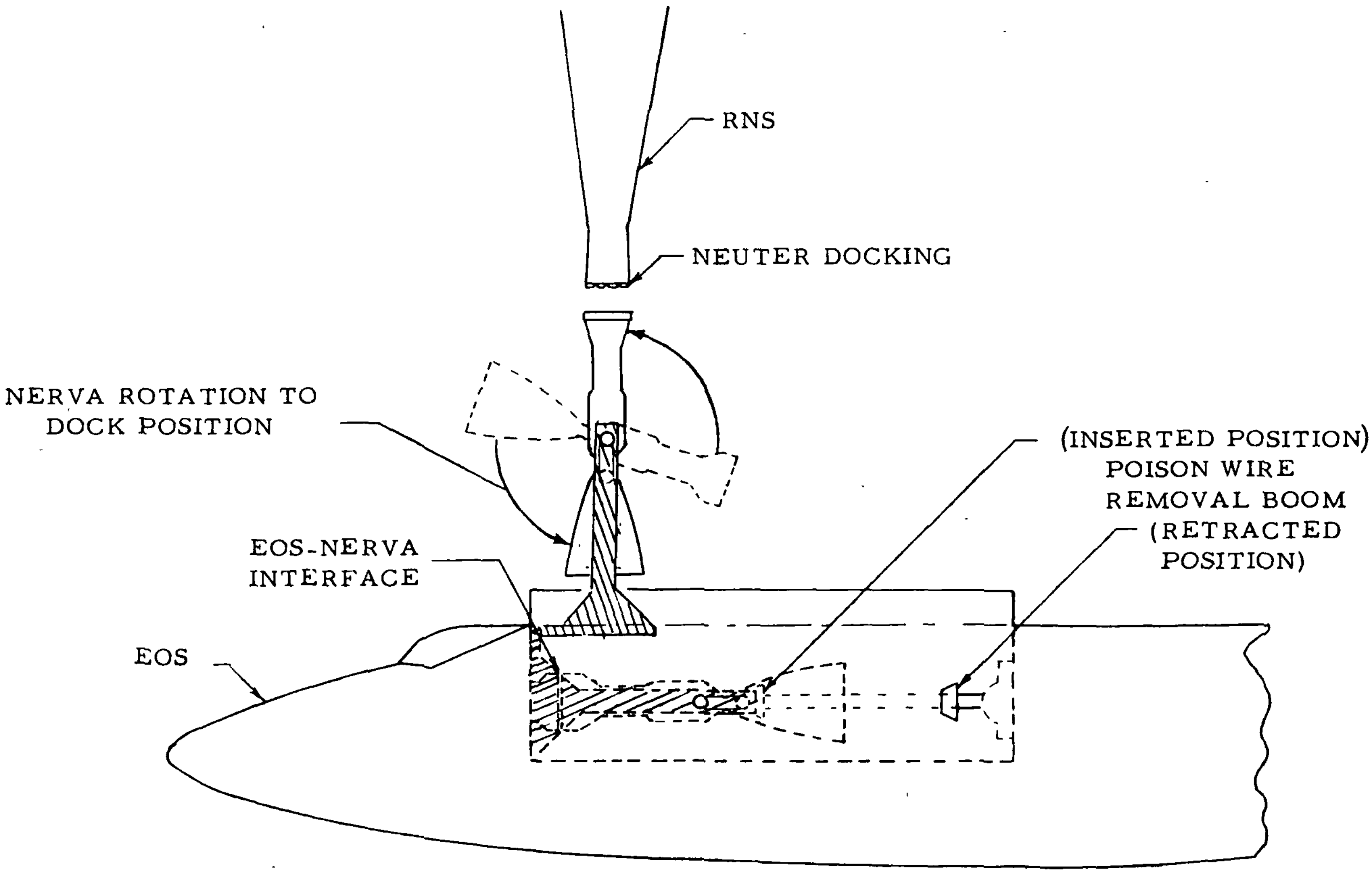

Figure 7-15 Poison Wire Removal and Engine Mating Concept

STAGE MAINTENANCE INTERFACE

Two maintenance element concepts have been identified for servicing the RNS in orbit. One concept is slaved to the Space Shuttle and is attractive for low traffic rates. It utilizes teleoperators under manual control from the space shuttle or the maintenance element. The other concept is 33 feet in diameter and is launched to orbit by the INT-21 where it remains stationed permanently. This would be desirable for high traffic rates.

Space Shuttle Launched Maintenance Element

The configuration of a Space Shuttle - integrated maintenance element developed for orbital servicing, replacement and checkout of the RNS equipment and components is shown in Figure 7-16. The element is designed as an integrated module, deliverable and deployable from the 15-foot diameter by 60-foot long cargo bay of the Space Shuttle, capable of all normal maintenance and replacement by remote controlled mechanical manipulation, and with a "cherry-picker" pod incorporated which can be manned to afford direct manual inspection and/or service of special case problems that might arise.

The chassis of the element consists of a 36-inch by 120-inch rectangular torque box with a neuter dock at each end, capable of manned occupancy or for through passage to the astrionics bay. The replacement components, manipulators, manned pod, etc., are located externally on the side of the chassis and confined within the 15-foot diameter envelope of the shuttle Remote control of the servicing operations may be conducted from inside the element chassis or from the Space Shuttle.

Three categories of RNS astrionics bay equipment require replacement: the RCS subsystem, which is internally packaged in two symmetrically located semicircular modules arranged to be removed as integral units; the internal electrical and electronic components, which are annularly installed on removable racks around the inside of the astrionics bay outer wall between the RCS modules; and the external sensors and antennas which are mounted in modular groups on the outside of the astrionics bay outer wall. The maintenance element is equipped with a pair of special manipulators to service the RCS modules, plus a general manipulator capable of all of the other servicing functions including emplacement of the manned pod.

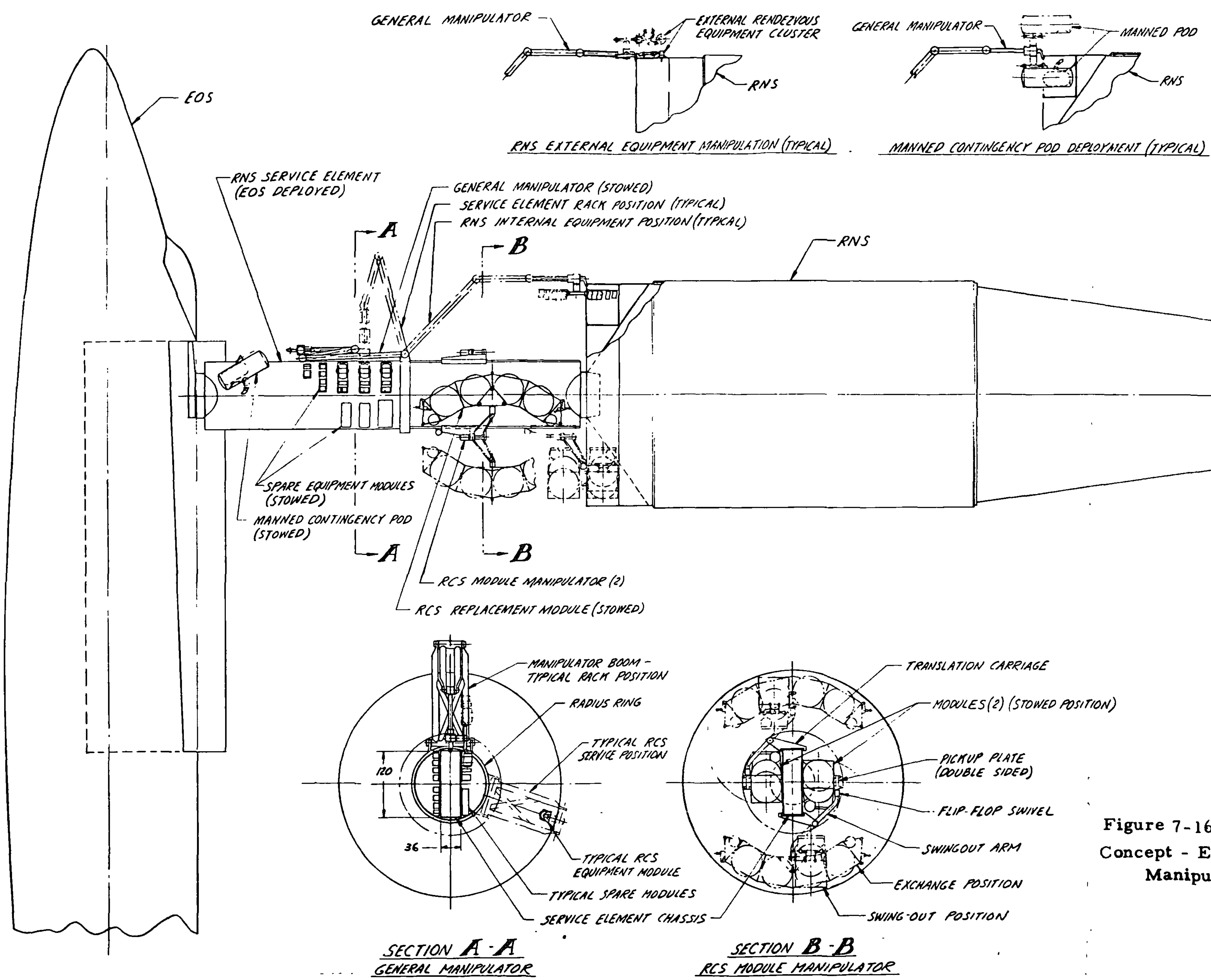

Figure 7-16 RNS Maintenance Element Concept - EOS Deployed/Mechanical Manipulator Implemented

The pair of RCS manipulators are located nearest the RNS end of the service element together with provisions for storage of a pair of RCS modules Each manipulator consists of a translating carriage railed to the service element chassis, a swing-out arm and flip-flop spindle with dual (opposed) RCS module pickups. In a typical service operation, two filled RCS modules are initially stowed on the maintenance element already engaged with one of the pickups of their respective manipulators. The following sequence is used to exchange the filled modules with the spent modules in the RNS: (1) the filled modules are swung out and rotated away from the RNS; (2) the manipulators are then translated to engage the unoccupied pickups with the RNS spent modules; (3) the spent modules are withdrawn and the flip-flop spindles reversed; (4) the filled modules are inserted, and (5) then the manipulators stow the spent modules in the maintenance element in the positions formerly occupied by the filled ones. Suitable guide rails and automatic couplings and latches are provided, and the protective covers over the RCS bottles integral with the modules, so that no other operations are required in making the exchange.

The general manipulator, together with a complete complement of spare equipment modules, are mounted behind those for the RCS system. The manipulator and spares are arranged so that any modular group of components in the RNS can be exchanged with a replacement from the maintenance element. The manipulator embodies a multiple articulated boom mounted off a ring around the maintenance element chassis and capable of all necessary general reaches and orientations, incorporating an anchor pad and translation/rotation head at the boom end to facilitate the local manipulations. A dual pickup flip-flop spindle in the head permits the exchange of a module with its replacement, similar to the RCS system scheme. The anchor pad on the end of the boom engages with points on the RNS and service element structures to provide suitable accuracy and rigidity in the pickup and emplacement of the modules. The offset between the anchor pad and the pickup allows engagement with either internal or external modules on the RNS or those on either side of the maintenance element chassis, from the same line of anchor points.

The manned pod is attached to the maintenance element chassis through an airlock for pressurized entry and exit; and is equipped with suitable connections so that it can be picked up by the general manipulator and transported to the vicinity of any of the equipment modules. By incorporating arm sockets and an external complement of tools, various special servicing and/or repair functions can be accomplished.

INT-21 Launched Maintenance Element

A 33-foot diameter maintenance element could facilitate the maintenance operations and although not compatible with the Space Shuttle launch, it can be boosted as an adjunct to the RNS tank on an INT-21. The configuration, as shown in Figure 7-17, is compatible with the RNS geometry with a single positioning. Inflatable seals would mate the docking and equipment bay structure enabling the equipment bay to be pressurized. In this manner, personnel would have direct access to the astrionic equipment without need for EVA. Also, replenishment of RCS and fuel cell reactant could be accomplished by a mechanism controlled from within the maintenance element and monitored through viewing ports.

The design approach provides for future growth through modular expansion. In addition, it can be employed to service other program elements such as space station or the propellant depot presently under evaluation. Also, the element could be transported to lunar orbit to service disabled RNS, space tug, or OLS, should the need arise.

Figures 7-18 and 7-19 illustrate a conceptual approach to replacement of the O_2/H_2 tank modules located in the astrionics bay. The approach is applicable to both the 33- and 15-foot diameter service element concepts outlined above. The spherical tanks supply propellant to the RCS as well as to the fuel cells used to generate electrical power. The total propellant is contained in eight tanks, six hydrogen and two oxygen. The system has been subdivided into two modules containing three hydrogen tanks, two oxygen tanks, and two quads of RCS thrusters. These modules are interconnected to maximize reliability by ascertaining that all propellant tanks are available to all RCS quads and fuel cells. Each module is about 25 feet long, 9 feet wide, and 6 feet deep.

In order to avoid the problem of removing a portion of the astrionics bay structure or disconnecting fluid and electrical line to the RCS thrusters when removing each module, the RCS thruster panels are hinged inboard prior to module extraction as shown in Figure 7-19. This is accomplished by an electrical rotary actuator at the panel hinge point. Swivel joints are featured on the hinge line to obviate flexible joints in the fluid lines, however, the electrical cabling will be flexible. Three handling hardpoints are provided at the top of each module penetrating the micrometeoroid and thermal cover. Three probes which are provided by the maintenance element latch mechanically into hardpoints. By retracting the probes simultaneously, each module is extracted from its compartment.

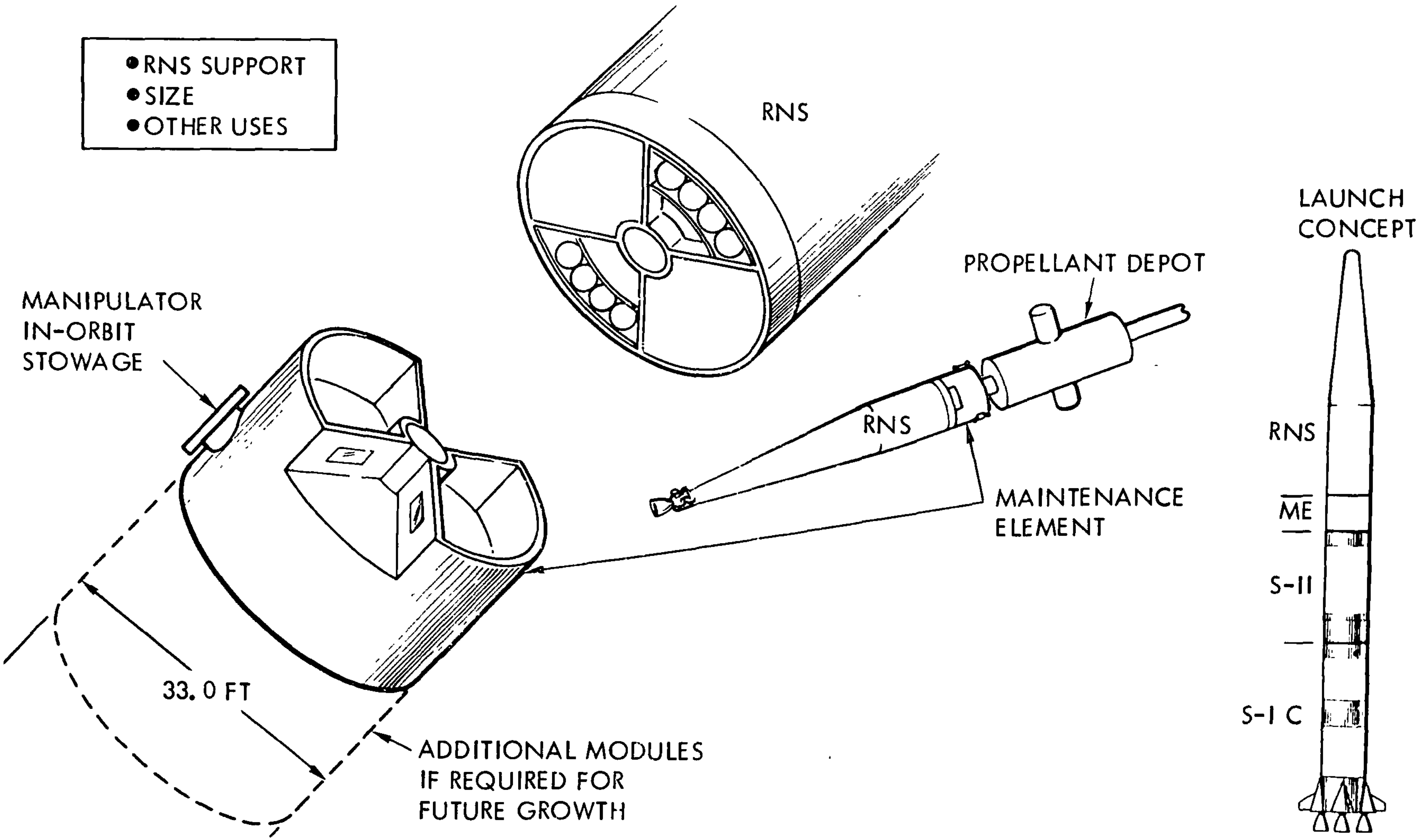

Figure 7-17 RNS Maintenance Element Concept - INT-21 Launched

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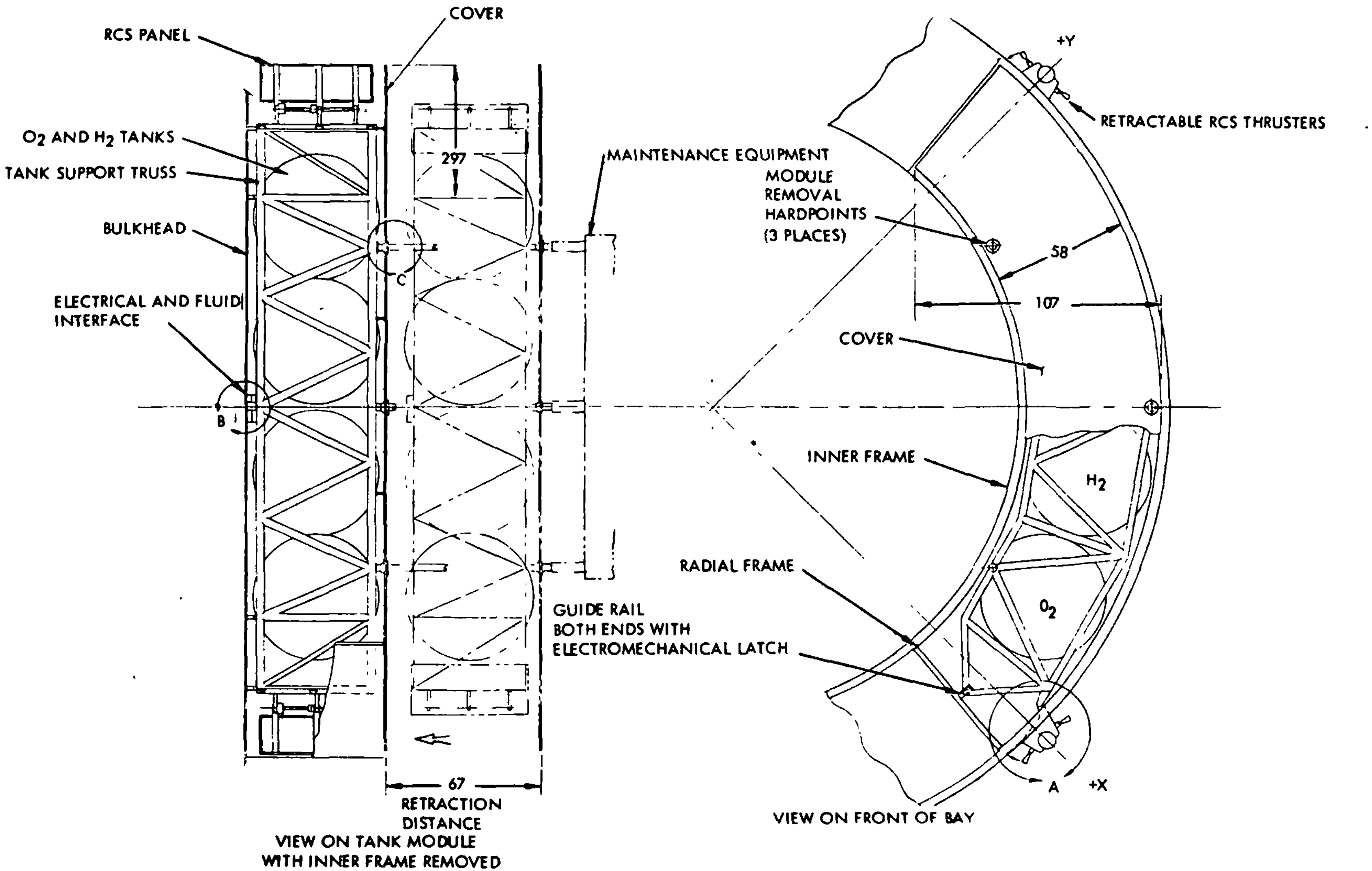

Figure 7-18 Tank Module Replacement Concept

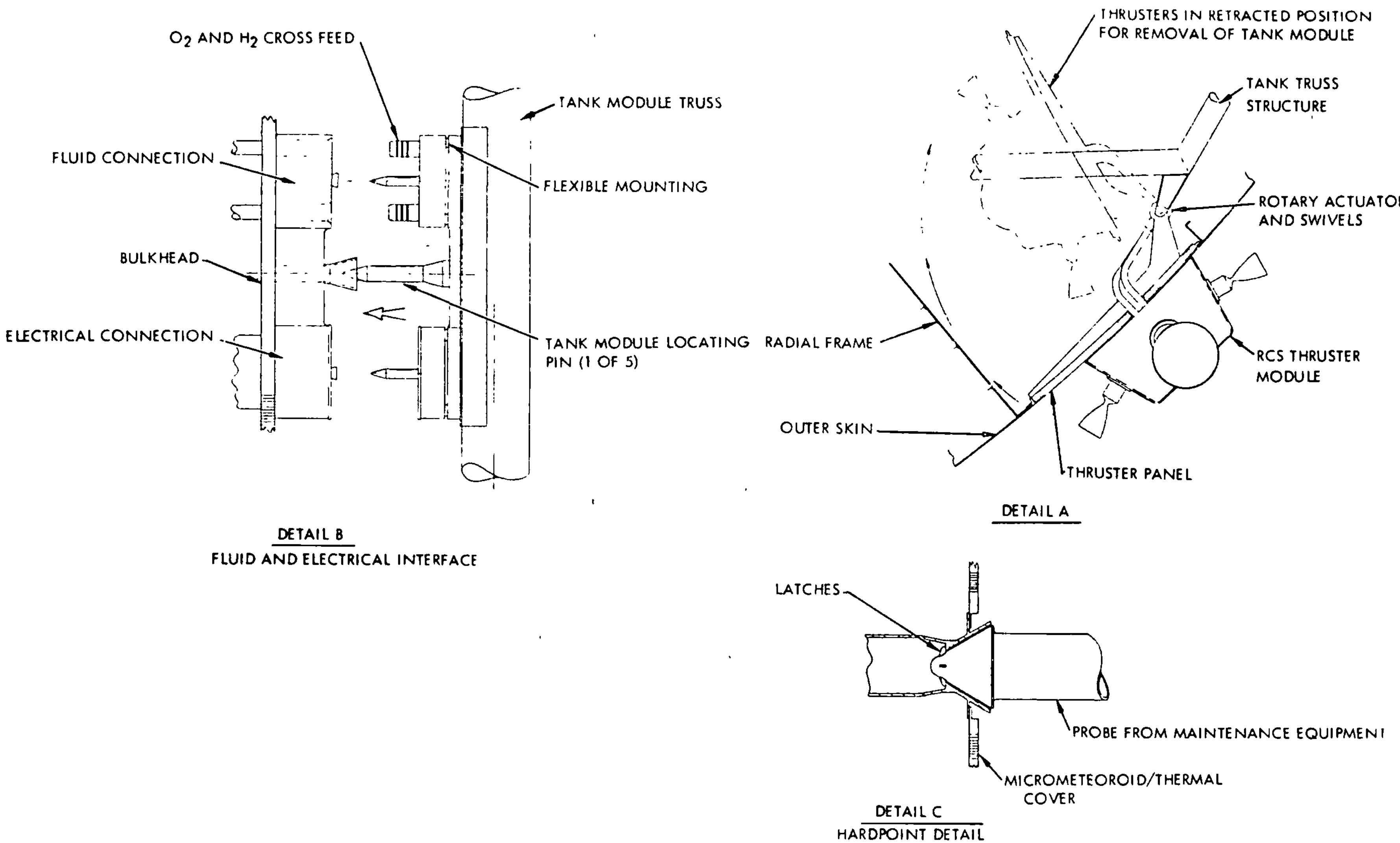

Figure 7-19 Tank Module Replacement Concept Details

To replace fully fueled modules into an empty compartment, the three probes are extended from the maintenance element over a distance of 67 inches. Two rails on radial frames with mating rollers on the module truss structure guide and position the module as it is lowered into the compartment. Final positioning is achieved by four tapered pins on each corner of the tank truss structure mating with corresponding sockets on the bulkhead. A fifth pin is required at the fluid electrical disconnect located on the centerline of the module.

The disconnect consists of two separate connectors, one for fluid and one for electrical lines. The connector half attached to the tank module is mounted semi-rigidly to take up any small variations in location which could exist during mating. Additional alignment pins are provided for each connector. After final positioning, the tank module is held in place by two latches on the guide rails.

Two components of the electrical power system require replacement during the RNS lifetime - the fuel cells and the batteries. The replacement concept utilizing the shirtsleeve environmental and manual servicing capability of the maintenance element concept are depicted in Figures 7-20 and 7-21. Each fuel cell and the battery rack are individually mounted and pinned to track sections which extend to the forward end of the stage and align with corresponding tracks in the service element when they are coupled. All system connections between the fuel cells and stage and between the battery pack and stage are collected into a single row on a disconnect panel attached to the inboard side of each element for ready access. All couplings are of the quick disconnect half-turn type, with self-sealing provisions in the line couplings. By manual disengagement of the couplings and track pins, a fuel cell or the battery rack can be controllably moved into the maintenance element, diverted to storage and a new unit emplaced. The provision for individual replacement of the fuel cells permits servicing without complete power shutdown of the RNS. For the same reason, it may be desirable to split the battery pack into two separately replaceable racks, which can readily be done. Other components of the electrical system are designed to operate for the life of the vehicle and hence are modularly installed with pullout trays to be serviced in place or removed as necessary, similarly to the other "full life" subsystems.

SPACE TUG INTERFACE

Prior discussion was oriented to maintenance operations in the forward end of the stage where radiation levels are low enough to permit direct manned repair and servicing operations. However, following NERVA firing, the radiation levels in the immediate vicinity of the engine will require the use of a remotely controlled manipulator such as an unmanned space tug.

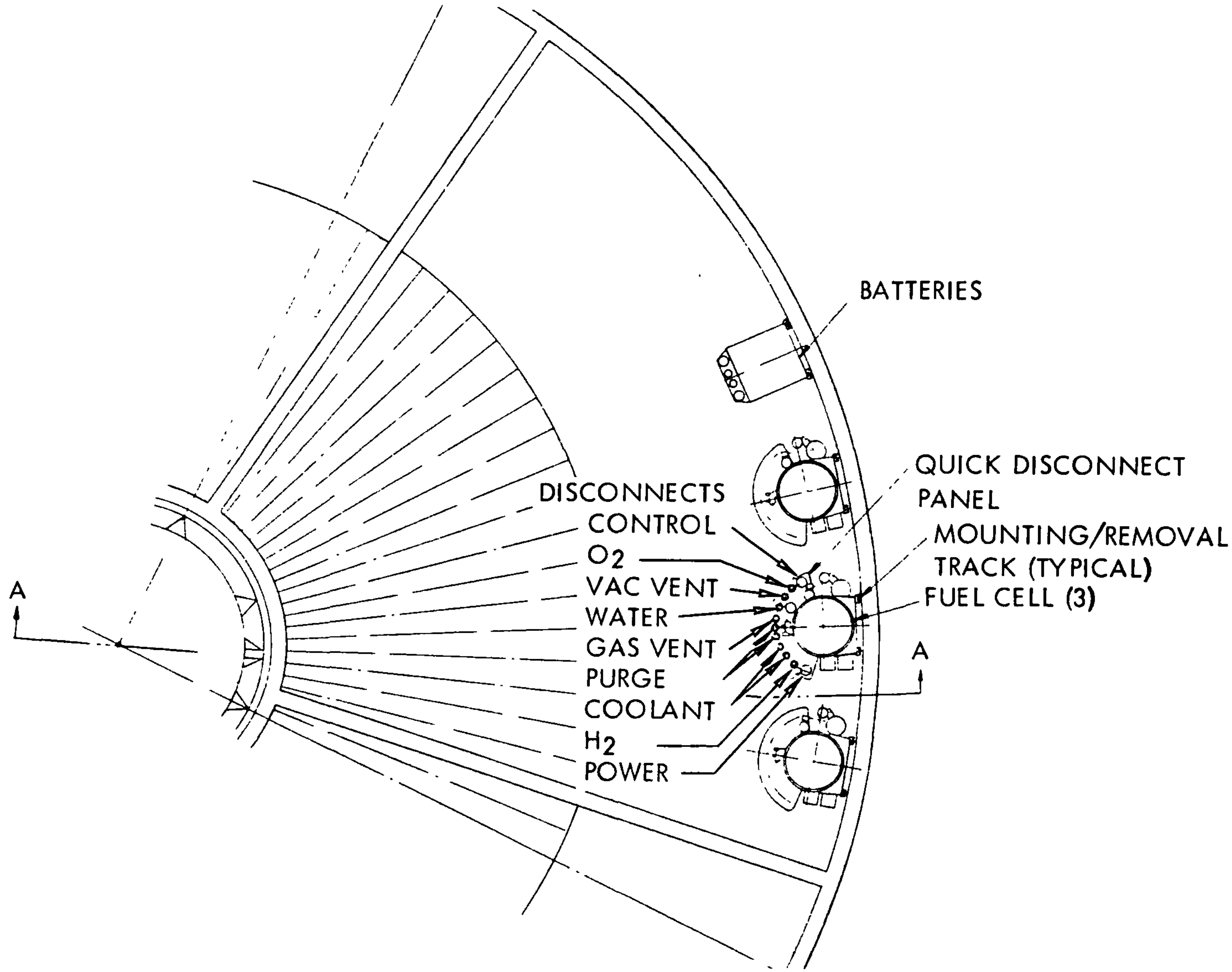

Figure 7-20 Electrical System Orbital Service Concept

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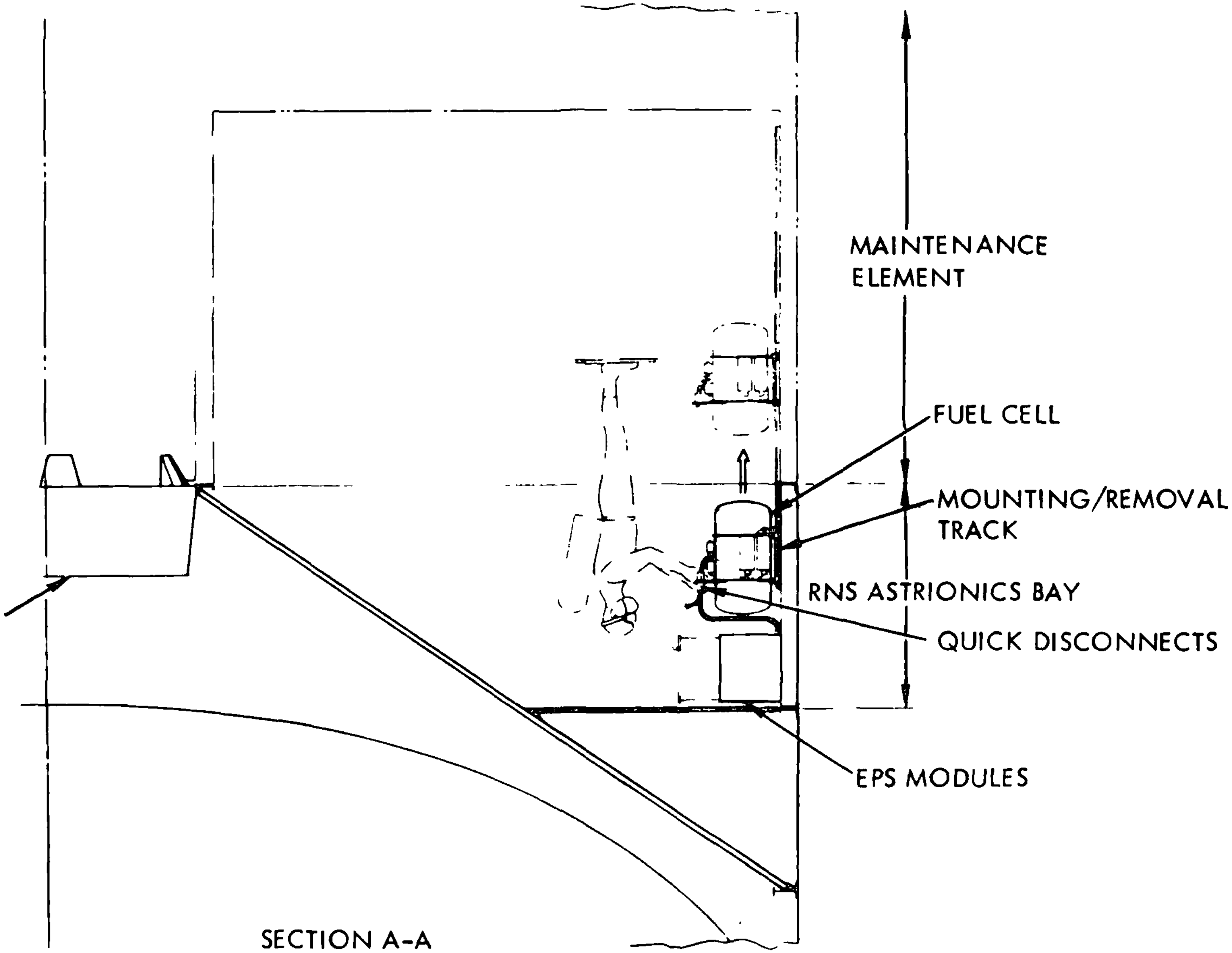

Figure 7-21 Electrical System Orbital Service Details

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The primary maintenance operation considered in this preliminary analysis consists of the removal and disposal of the NERVA. The removal will be initiated with disengagement of fluid and electrical connections at the docking system interface plate. This will be followed by extending and unlocking the docking rings and positioning the engine for mating with the tug's acquisition probe as shown in Figure 7-22. The tug would then maneuver to a position for engaging its extensible acquisition probe with the engine passive docking ring. After lock between the two systems is verified, the RNS engine manipulator would be released, permitting engine removal and disposal as required. The mechanism shown in Detail "A" of Figure 7-22 is normally stowed. It is activated for engine acquisition and removal by rotating about its hinge point locked to the side of the thrust structure by bolts emplacement struts. Latching and unlatching of the engine is accomplished as shown in the figure. The rotary actuator for the extension of the engine separation strut is shown in Section B-B.

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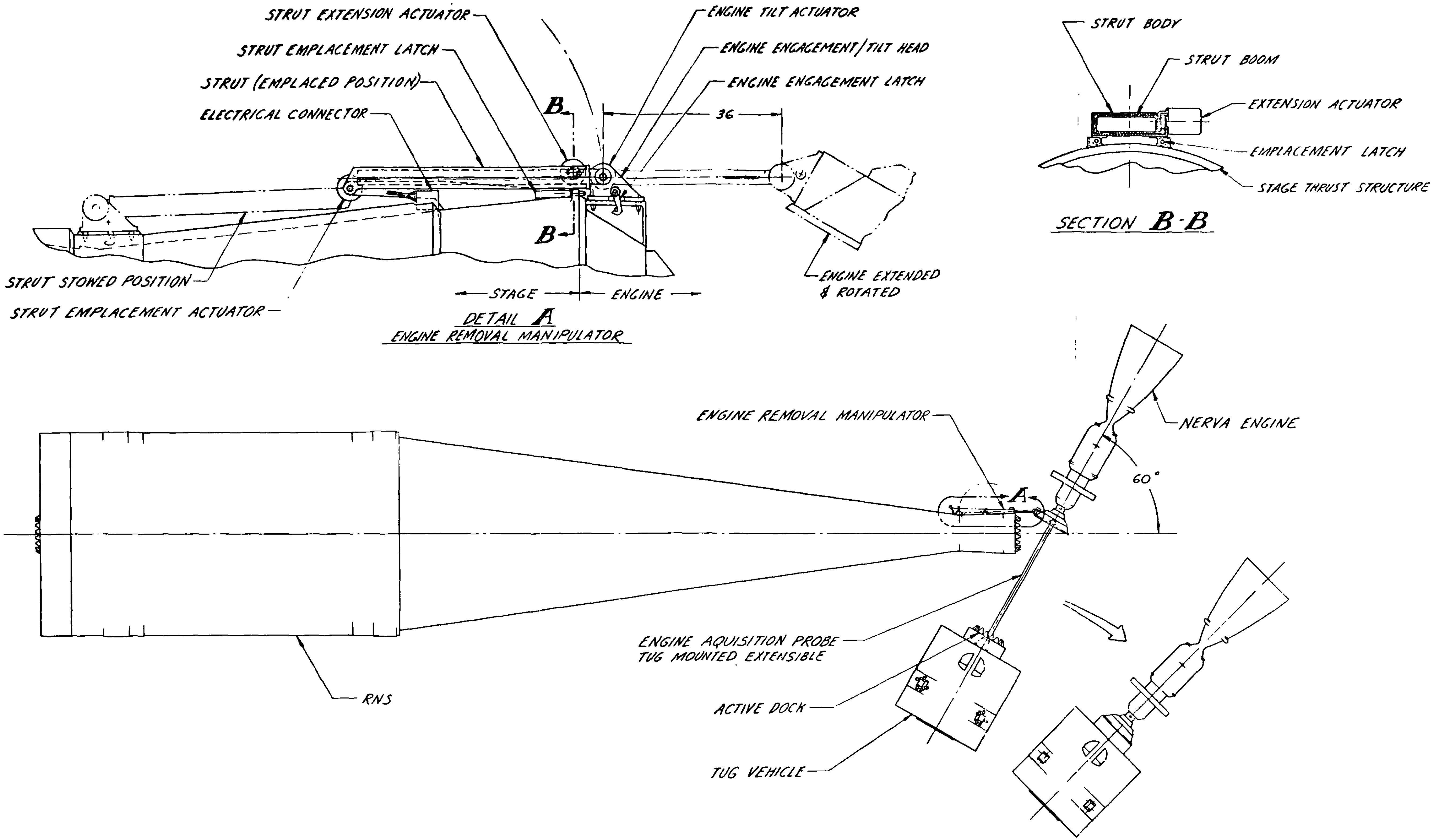

Figure 7-22. Concept - RNS NERVA Engine Removal

8.0 INT-21 INTERFACE

The primary structural, mechanical, and subsystem interfaces between the RNS and the INT-21 are presented in Figure 8-1.

The major interface which exists between the two stages occurs at station 2519, the details of which are shown in detail A, and sections C and D. Detail A defines the clearance envelope at the center of the S-II stage at the interface plane which must be honored by the S-II stage in order to clear the RNS neuter dock. The clearance envelope precludes any S-II hardware extending into the specified area when the S-II is in the pressurized condition.

Sections C and D define the physical location of the three alignment guide pins which are on the RNS adapter structure and the mating guide pin receptacles attached to the S-II forward skirt. Specific details of the guide pins and receptacles are shown in section G.

Structural attachment between the S-II and the RNS at the station 2519 interface plane is accomplished by a circumferential pattern of 256 high-strength bolts. The installation of the bolts and physical characteristics of the mating ring frames are shown in section H.

A major system interface which exists between the S-II and RNS occurs at the RNS adapter flight separation plane and is illustrated in sections K, B, and view J. This interface consists of eight electrical quick-disconnects integrated into a panel assembly shown in section B and view J. This system is basically similar to that which exists between the S-IVB and S-II stages.

Detail F illustrates a typical section of the S-II forward skirt structure which consists of an outer skirt stiffened by forward hat sections and circumferential frames. A typical physical location of one of the guide pin receptacle fittings is shown.

The representative structural details of the RNS adapter are shown in detail E. This structure consists of an outer skin with integrally stiffened stringers, and circumferential frames. Dimensional details of the bracket which positions the indexing guide pin are also shown.

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Figure 8-1. RNS/INT-21 Interface

9.0 DESIGN CRITERIA AND CONSTRAINTS

INTRODUCTION

The design criteria and constraints were assembled and derived based upon mission operations, environment to which the nuclear stage would be exposed, its interfaces with the Earth launch vehicle, constraints imposed by facilities, design assurance, and safety considerations. The study effort on Reusable Nuclear Shuttle concentrated on the lunar shuttle and the synchronous orbit shuttle missions.

The criteria and constraints were divided into eight categories: (1) Mission Operations, (2) Environment, (3) Interfaces, (4) Facilities, (5) Fabrication and Materials, (6) Transportation and Storage, (7) Safety and (8) Structure.

Certain documents are referred to where data are so voluminous that it is not practical to include in this report. Data frequently used in design or analysis of the RNS were extracted from source documentation for ready reference.

MISSION OPERATIONS

The RNS must be capable of performing interorbital transfers and unmanned planetary injection missions in accordance with the guidelines of Reference 9.1. The RNS will ultimately be used for manned planetary missions. Interorbit transfer missions will be comprised of shuttles between low Earth orbit and lunar or synchronous orbit. Retrieval of the RNA will be accomplished on the planetary injection mission unless the RNS has reached its end-of-life span. The lifetime design goal for the RNS will be three years in space with the capability for maintenance in Earth orbit. In-orbit maintenance and propellant refueling will be accomplished only at the RNS operations orbit defined as 260 n mi circular at an inclination of 31.5 degrees. The stage will be checked out in the RNS operations orbit prior to each mission.

The RNS must be capable of performing both manned and unmanned shuttle missions. Stay time in lunar or synchronous orbit will not exceed 30 days; however, the RNS cycle time can extend to 54.6 days taking into account transit and turnaround time.

* The minimum performance of the RNS for the Lunar Shuttle Mission utilizing the orbital start mode is TBD pounds delivered to lunar orbit with 20,000 pound payload returned to Earth orbit.

** The minimum performance of the RNS for the Synchronous Orbit Shuttle Mission is TBD pounds delivered to synchronous orbit and 20,000 pound payload returned to low Earth orbit (270 n mi).

Guidelines for the Manned Planetary Mission Spacecraft for the opposition class with a Venus swingby and the conjunction class Mars missions are summarized below:

	Opposition Class(4)	Conjunction Class(5)
Planetary Mission Module(1)	145,000 lb	170,000 lb
Manned Mars Excursion Module(2)	100,000	200,000(6)
Probes at: Mars(2)	30,000	30,000
Venus	6,000	-
MEM & Probe Compartment(2)	5,500	11,500
Overboard Expendable Rate(3)	13 lb/day	13 lb/day

Notes:
(1) PMM is retained throughout the mission
(2) Jettisoned at Mars
(3) Includes attitude control, atmospheric leaks, etc.
(4) 560 day mission
(5) 1040 day mission
(6) Two MEM's are included for conjunction class missions

The general weight variation for the PMM is determined by:

$$PMM = 117,292 + 50.73\ T$$

where T is total mission duration.

---

* Minimum lunar mission performance in Phase I (Reference 9.2) were identified as:

1. 119,000 pounds delivered to lunar orbit with zero payload return.
2. 44,000 pounds delivered to lunar orbit and 44,000 pounds returned to Earth orbit.
3. Zero payload delivered to lunar orbit and 69,000 pounds returned to Earth orbit.

** Minimum synchronous orbit shuttle performance in Phase I were identified as:

1. 102,000 pounds delivered to synchronous orbit and zero payload returned to low Earth orbit.
2. 38,000 pounds delivered to synchronous orbit and 38,000 pounds returned to low Earth orbit.
3. Zero pounds delivered to synchronous Earth orbit and 60,000 pounds returned to low Earth orbit.

Orbiting Lunar Station (OLS) will have a 60 n mi circular orbit inclined 90 degrees to the equator. The maximum RNS stay time in lunar orbit will not exceed thirty days. A safe separation distance in same orbit will be maintained between the RNS and the OLS. A Space Tug will transfer payload modules between the RNS and orbiting station. When extended stay times in lunar orbit are required, the RNS may be maneuvered to a point in orbit where minimum RNS attitude controls are required from a radiation dose to OLS crew standpoint. A safe distance will be maintained between the RNS and OLS during rendezvous and departure (10 n mi orbit separation, i. e., 60, and 70 n mi orbits).

The Geosynchronous Orbit Station (GOS) will be located in a geosynchronous orbit. Maximum RNS stay time in synchronous orbit will not exceed thirty days. A safe separation (in same orbit) will be maintained between the RNS and the GOS. Payload transfer between RNS and GOS will be accomplished with a Space Tug. A safe distance will be maintained between the RNS and GOS during rendezvous and departure (10 n mi orbit differential).

A number of shuttle missions are specified as unmanned missions which necessitates the incorporation of automatic rendezvous and docking capability in the RNS design. The minimum separation distance (in same orbit) for the OLS or GOS is TBD n mi and the RNS must be capable of rendezvous to this position and hold until payload transfer is accomplished by Space Tug. The automatic rendezvous is required at the Propellant Depot (PD) as well; however, docking the RNS to the PD may be accomplished by tug and/or PD crew through RF link with the RNS.

At the present time, the RNS is started only after being injected into orbit by an INT-21 launch vehicle. The INT-21 with J-2 engines has the capability to launch approximately 265,000 pounds to a 100 n mi orbit with a 30-degree inclination or 190,000 pounds to 260 n mi orbit with a 30-degree inclination as indicated in Reference 9.3. RNS operational interfaces for lunar shuttle, synchronous orbit shuttle, and injection missions are illustrated in Figure 9-1.

Initial use of the RNS is restricted to single module applications to shuttle and injection missions. A design objective of the RNS is the ability to evolve into a design which permits multiple RNS module application to advanced manned planetary missions. In this instance, mission duration of two years and longer, no expendables replenishment can be anticipated for the module utilized on the return leg from the planetary mission.

Tye 60-cycle limit (set by Reference 9.4) of NERVA engine operation at full rated conditions must be taken into consideration when developing the traffic model since engine burns can vary from four to nine, depending on

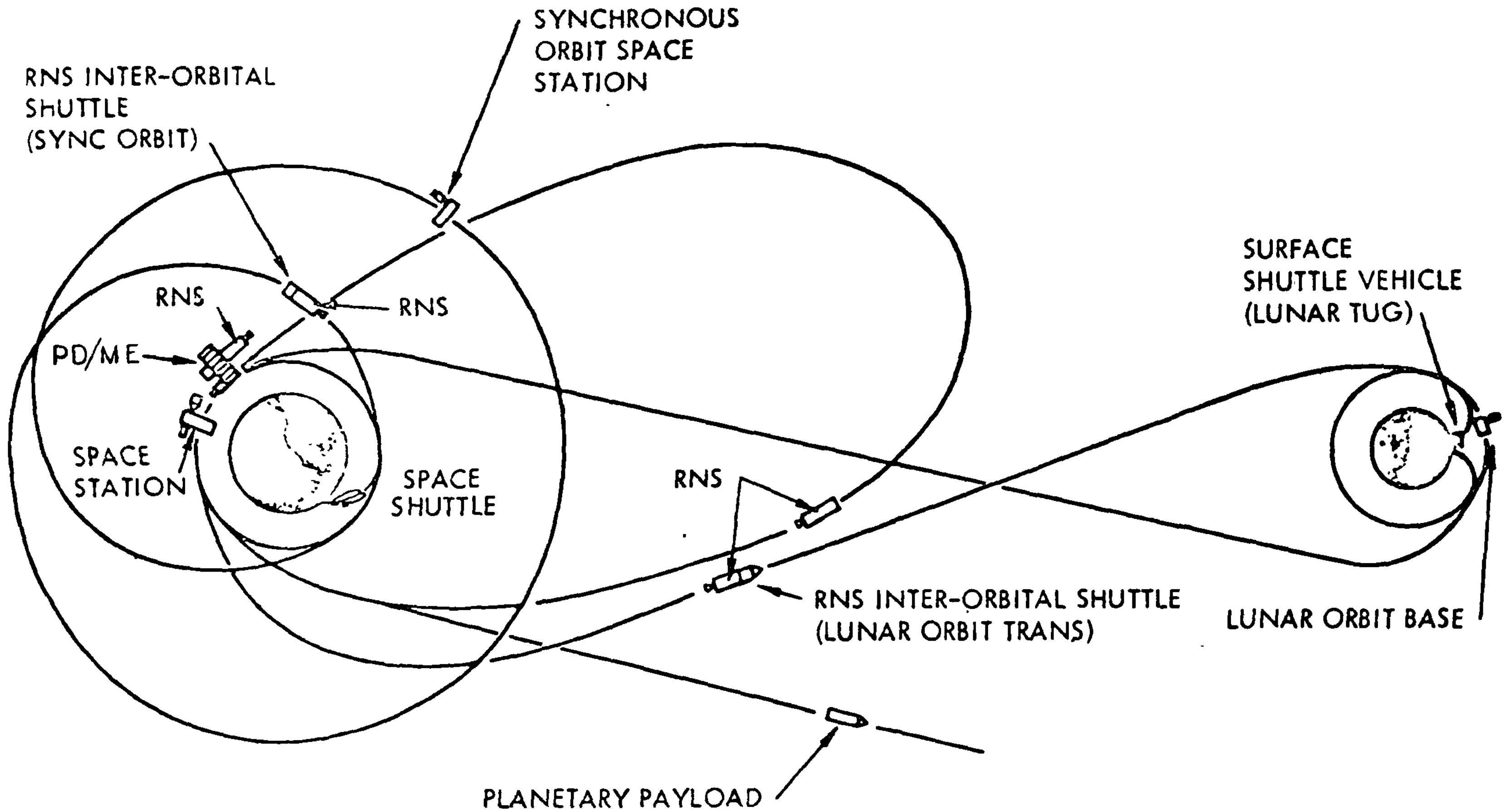

Figure 9-1 Operational Interfaces

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mission being flown. The end-of-life burn for spent stage disposal is also to be included to stay within the 60-cycle limit. Disposal of a spent RNS will be made to a safe long life Earth orbit or a heliocentric orbit. Only in the case of emergencies will disposal in deep ocean waters be considered.

The study guideline establishes criteria the RNS stage must adhere to, and are identified in the following discussion. The RNS configuration is a 33-foot diameter "single tank" configuration that is launched integrally to orbit by a Saturn V INT-21 vehicle. Baseline RNS configuration shown in Figure 9-2 will require the use of the Space Shuttle as well as the INT-21 launch vehicle to launch the RNS to the operations orbit. The RNS propellant tank will be launched by the INT-21 launch vehicle and the propulsion module (NERVA engine plus auxiliary tank or NERVA engine alone) will be launched by the Space Shuttle. The layout of the selected baseline configuration shows the general arrangement of the vehicle. The single cell tank of 33-foot cylindrical diameter and 8-degree half cone angle/25-inch cap radius aft bulkhead is 1827 inches long and accommodates 300,000 lbs of propellant with 5% ullage. Extension skirts are added to the forward and aft ends of the cylindrical portion for attachments of launch vehicle/payload and aerodynamic shroud, respectively. A thrust structure, also in the form of a skirt, is provided at the end of the aft bulkhead to accommodate the engine installation. An active neuter docking system and supporting cone structure are incorporated in the forward skirt for stage docking to a propellant depot or similar facility and for payload module connection. An annular astrionics bay for installation of the stage auxiliary subsystems is also integrated into the forward skirt structure. Another active neuter docking system is built into the thrust structure to facilitate orbital installation of the engine as well as orbital handling of the stage and the NERVA engine for removal and disposal operations. The passive assembly of the neuter dock is attached to the engine forward thrust plate and the active assembly is attached to the stage thrust structure. The tank is compartmented by three perforated capillary barriers and a bottom screen to facilitate propellant location management.

Initial design concepts will reflect a 1974 state-of-the-art with consideration given to later incorporation of more advanced technology. All versions of the RNS will be man rated; i.e., they will meet all structural, materials, and quality standards required for manned application. Provisions will be made in the RNS for manual override control by the manned payloads on manned shuttle missions.

Design criteria generated by orbital operations are covered in the following discussion. Logistics operations prior to embarking on a shuttle run include preparation of the stage (post boost) after initial injection to orbit, performance of maintenance operations, loading of propellants, mating of payload to RNS and, finally, checking stage/payload after mating. RNS designs

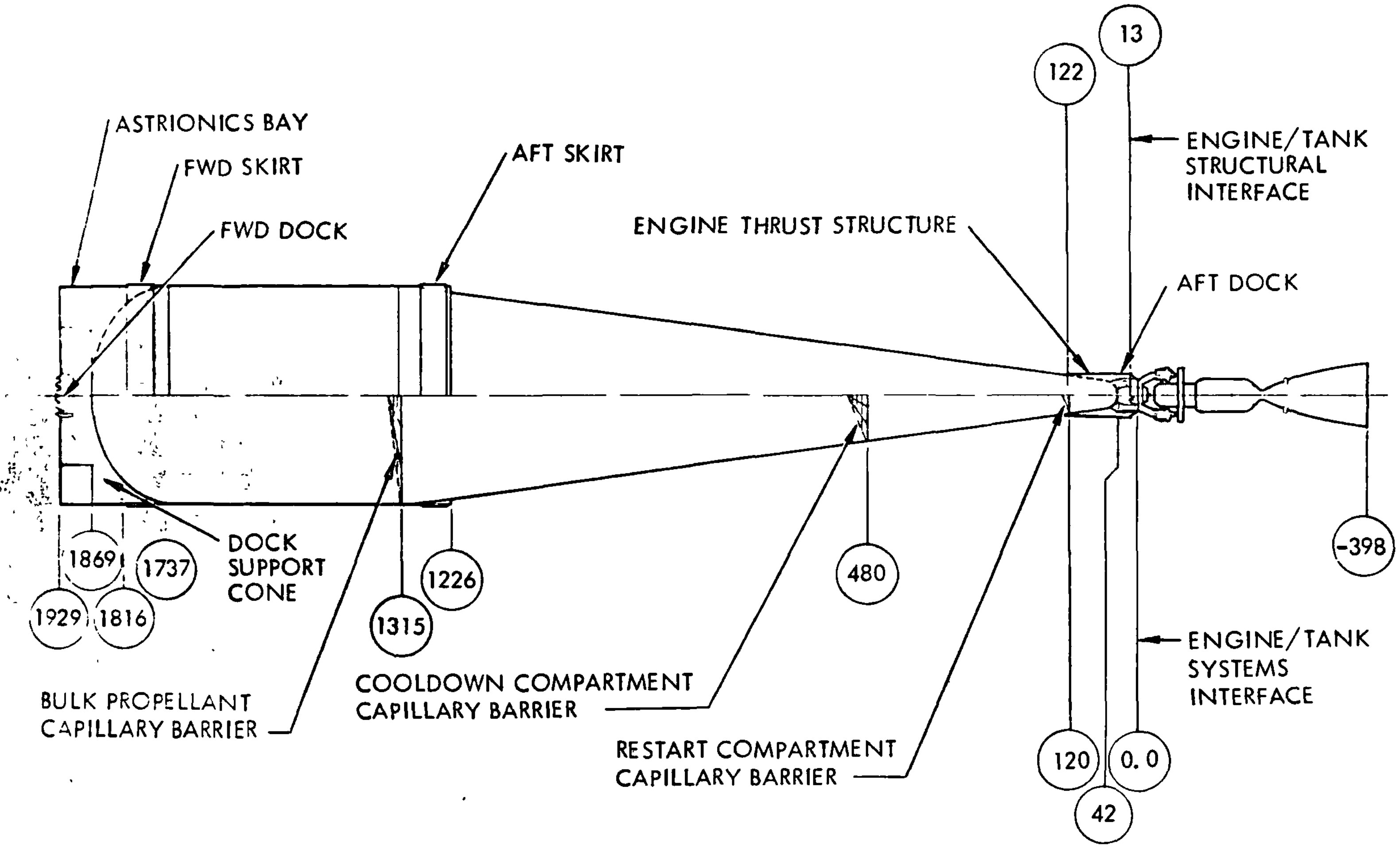

Figure 9-2 Selected RNS Baseline Layout

must provide for removal of safety devices and shrounds required for boost to orbit. Assembly and maintenance operations will require the use of Tug with manipulators and/or limited extravehicular activities (EVA). In order to accomplish the orbital assembly and maintenance functions, system elements or assemblies should be packaged to facilitate removal and replacement by either remote manipulators or service crew faced with the operational constraints of space environment. Flight Replaceable Units (FRU) should be made accessible such that removal of one FRU does not first require the removal of another FRU. Since removal and replacement activities will be accomplished in the zero "g" environment, provisions will be made for purchase or anchor points for personnel or manipulator performing the removal and replacement operation. The FRU's are to be designed with release and locking mechanisms to be compatible with the limitations of remote manipulators.

Preliminary maintainability design criteria to be utilized on the RNS is listed below.

1. Subsystem or component failure shall not propagate sequentially; equipment shall be designed to fail operational/fail operation/fail safe. (Recent studies indicate fail operational/fail safe may be adequate.)

2. High energy release equipment, such as pressurized tanks, propellants, etc., shall be located or protected so that a failure of one will not propagate to others or cause catastrophic damage.

3. Replaceable unit design shall permit direct visual and physical access with connectors and couplings for ease of removal/replacement. The requirement for precision assembly of elements shall be avoided where possible and, where necessary, be provided with suitable guides and locking devices to aid in replacement. A minimum of tool and particularly special tool requirements shall be a goal in designing/selecting interfaces for replaceable units.

4. For those malfunctions and/or hazards which may result in time-critical emergencies, provision shall be made for the automatic switching to operation/safe mode.

5. Where feasible, electrical equipment shall be designed to be electrically isolated by interlocking switches or equivalent before physical access to exposed hazardous connections and compartments is possible.

6. Where possible, electrical and electronic spacecraft devices shall incorporate protection against reverse polarity and/or other creditable improper electrical inputs during qualification, acceptance, and checkout tests, if such inputs could cause mission significant damage to the devices that would not be immediately and unmistakably apparent.

7. Servicing and test parts, not required to function in orbit, shall be designed to preclude leakage in orbit by removing or capping immediately after final ground use.

8. All systems shall be provided with adequate deactivation and monitoring capability to verify deactivation is sufficient to prevent personal injury or equipment damage during maintenance activity (i. e., pressure down, voltage off, mixture noncombustible, etc.).

9. Quick disconnects with self-sealing capabilities shall be considered for all fluid systems components susceptible to replacement, to facilitate credible on-line replacement and to prevent contamination.

10. Design shall provide for isolation of anomalies such that a faulty subsystem or replaceable unit may be deactivated either automatically or without disrupting its own or other subsystems.

11. Dead facing techniques shall be employed wherever possible, particularly in essential systems to permit required maintenance effort without subsystem shutdown.

12. Automatic on-board checkout and status monitoring capability shall include appropriate transducers and electrical, mechanical, pneumatic and hydraulic systems for performance verification and malfunction detection to the lowest level replaceable unit.

13. Limited life components shall be located so they are particularly accessible for easy replacement and retest.

14. Nonstructural replaceable units shall not include any portion of the vehicle's integral structure, nor require any removal or disturbance of this integral structure.

15. Commonality should be emphasized in the selection of power supplies and connectors.

16. Provide direct access to modules to be removed and replaced without removing other modules or hardware.

17. Thermal insulation should be designed and installed in panels that can be removed/replaced, with minimum temperature bonding requirements.

18. Certain structural or mechanical installations must be instrumented for Non-Destructive Testing (NDT) when visibility and/or accessibility is restricted. Methods of NDT include fiber optics, ultrasonic and piezo-electric coating. Same readouts may be a part of On-Board Checkout (OBCO).

19. The on-board checkout assembly shall be capable of checking out the operation of a subsystem following maintenance without inducing failures leading to hazardous situations if the maintenance action has been incorrectly performed or the subsystem malfunctions in a credible manner.

20. Interlocks, automatic valves, or other means of ready isolation shall be provided for liquid and gas systems to preclude inadvertent leakage during maintenance efforts.

21. Design equipment and components which are subject to maintenance to preclude improper installation or connection.

22. Design to require the minimum personnel skills and training needed to develop adequate maintenance proficiency.

The RNS will require on-board checkout equipment to meet the autonomous requirement of having the astrionics system independent of the payload and performing a final automatic checkout of the vehicle after the payload is mated to the stage. In order to support the orbital maintenance operation, the on-board checkout equipment must detect malfunctions and isolate any fault to the FRU. Malfunction detection is not restricted only to the checkout in RNS operations orbit, but will be required throughout the entire mission. A final automatic checkout of the RNS/payload will be made in operations orbit prior to each mission.

Preliminary criteria to be utilized in developing on-board checkout system (OBCO) concepts are listed below:

1. Checkout of the RNS stage will be accomplished in the 260 n mi circular operations orbit while docked to the maintenance element and again after the RNS is separated from the maintenance element and mated to the payload.

2. RNS stage system operational status will be verified prior to departing lunar or synchronous orbit.

3. On-board automatic checkout will perform qualitative checkout (GO, NO-GO) used with predetermined parametric limits.

4. The OBCO will have self-check capability.

5. Operational signals rather than special stimuli will be used for checkout whenever possible.

6. Where a FRU cannot be operated during checkout because of operational or safety constraints, the OBCO will have the capability of checking it to the lowest possible level without actuating the output.

7. All redundant paths will be verified where redundancy is used internal to a FRU.

8. The OBCO will provide the means of recording malfunction detection data to assist in failure analysis at the most opportune time in the mission.

9. Malfunction isolation will be carried to the FRU level only.

10. The OBCO will provide checkout capability which is independent of ground participation.

11. Trend data analysis for failure prediction may be accomplished by support facilities, i.e., ground or orbiting space station/base.

12. Provide radio frequency (RF) and direct link for initiating checkout and transmitting checkout data to the space program element initiating the checkout.

Payloads, LH_2 propellants, and maintenance supplies for the RNS will be delivered by the logistics vehicle and assembled in the RNS operations orbit. The cargo bay of the Space Shuttle will be sized to have a clear volume of 15-foot diameter by 60-foot length. FRU sizes will be constrained by these dimensions. The round trip Space Shuttle payload capability was obtained from Reference 9.1. A payload of approximately 33,000 pounds can be delivered to the RNS operations orbit or approximately 44,000 pounds to the 100 n mi orbit inclined at 31.5°. In the latter case, a Tug would be required for transfer of the payload, brought up by the Space Shuttle, from the 100 n mi orbit to the RNS operations orbit.

ENVIRONMENT

Terrestrial Climatic

NASA TMX-53872, "Terrestrial Environment (Climatic) Criteria Guidelines for Use in Space Vehicle Development," 1969 revision, (Reference 9.5) shall be used as the model for climatic conditions the RNS will be exposed to at the various fabrication, test, and launch locations. The stage shall be designed to withstand these environments during test at the test sites and during launch from KSC. Protective equipment shall be provided for the RNS during transportation and storage to protect it from environmental conditions in which it was not intended to operate.

Figure 9-3 identifies the main geographical areas covered in the NASA TMX-53872 document.

Space Environment

NASA TMX-53957, "Space Environmental Criteria Guidelines for Use in Space Vehicle Development," dated September 1969 and NASA TMX-53865, "Natural Environment Criteria for the NASA Space Station" (Reference 9.6 and 9.7) shall be utilized as the environmental model for interplanetary space, terrestrial space, cislunar space, lunar, and planetary environments. Subject matter covered in the document include the following: definition of area of influence, meteoroid environment, radiation environment (solar high energy particle) has properties, radiation (thermal), magnetic field, wind regimes, ionosphere, solar particle prediction, and gravitational data.

Induced Environments

In addition to the natural environments discussed in the preceding paragraphs, the RNS will be subjected to induced environments caused by boost to orbit and NERVA engine run. These induced environments will include vibration, aerodynamic loading, aerodynamic heating during boost, and nuclear radiation during and subsequent to the NERVA engine run.

The nature and magnitude of induced environments which the RNS will be exposed to during boost to orbit can be found in Reference (8).

Recommended design acceleration for the NERVA during boost within the Space Shuttle Orbiter cargo bay is 3.0 g in any direction. Acoustic and lateral dynamics are TBD.

The induced radiation environment caused by NERVA engine run is mission dependent. The neutron and gamma radiation iso-contour levels are shown in Figure 4. Integrated flux or radiation levels may be obtained by multiplying the neutron flux densities or gamma kerma rates by engine run time.

The after-shutdown gamma kerma rates (rad-hr^{-1}) of Earth orbit insertion engine burn (EOI) for a representative reference mission are shown in Figure 9-5. The rates are given at a separation distance of 20 feet from the NERVA core center with reference to polar angles of orientation, $= 0°$ (vertical axis, toward direction), for varying after-shutdown decay times of 8 hours, shows the gamma kerma rates for after-shutdown decay times of 8 hours, 24 hours, 1 week, and 1 month. Gamma kerma rates for a nominal engine operating power level of 1575 mw illustrates that the dose rates are reduced by about 3.5 to 4 orders of magnitude during the first 8-hour post-shutdown period. Thereafter, incremental reductions by factors of about 4, 9, and

Figure 9-3. Main Geographical Areas Covered in Document

• UNPERTURBED NEUTRON FLUX DENSITIES AND GAMMA KERMA RATES

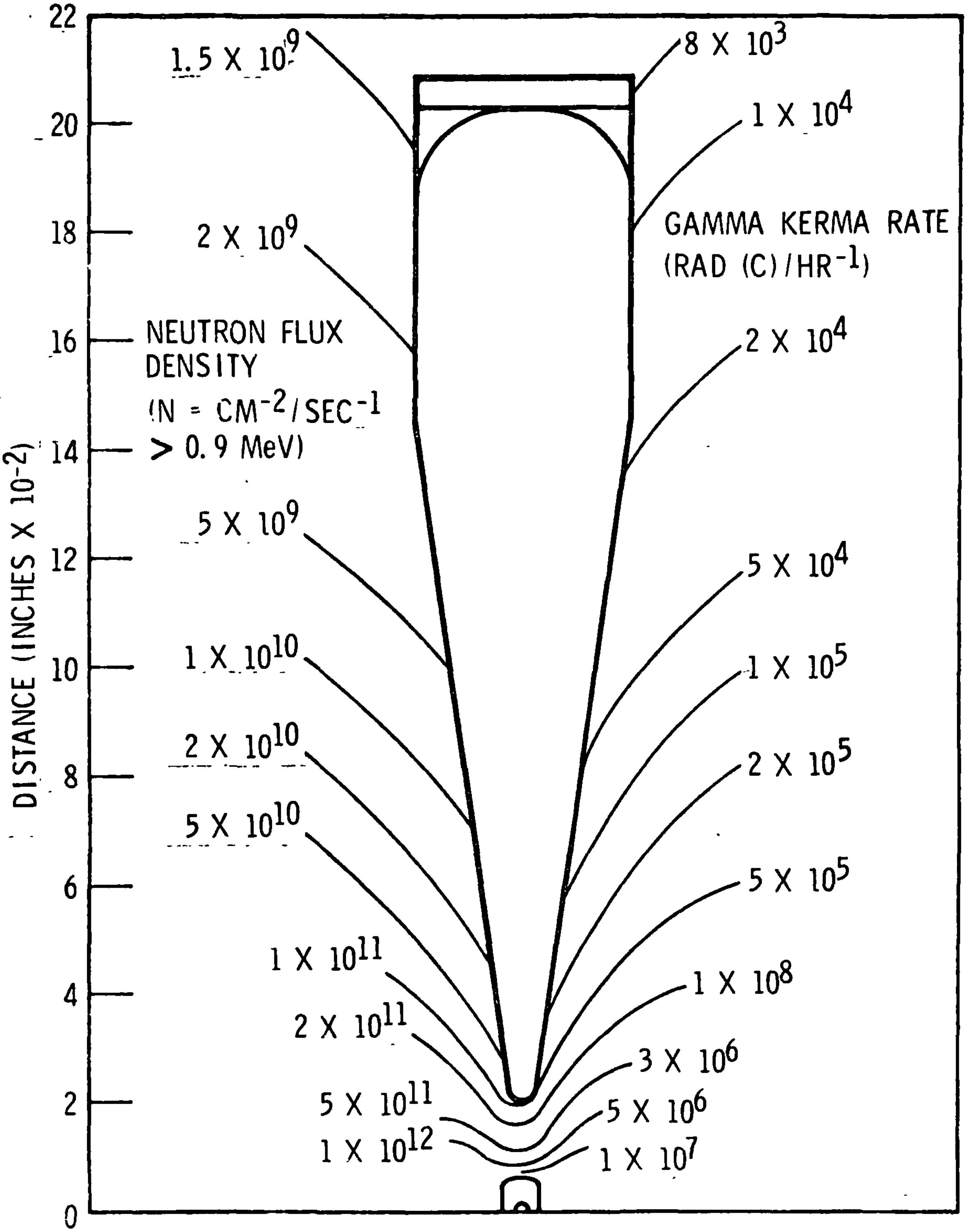

Figure 9-4 Iso-Dose Contours - RNS Baseline Configuration

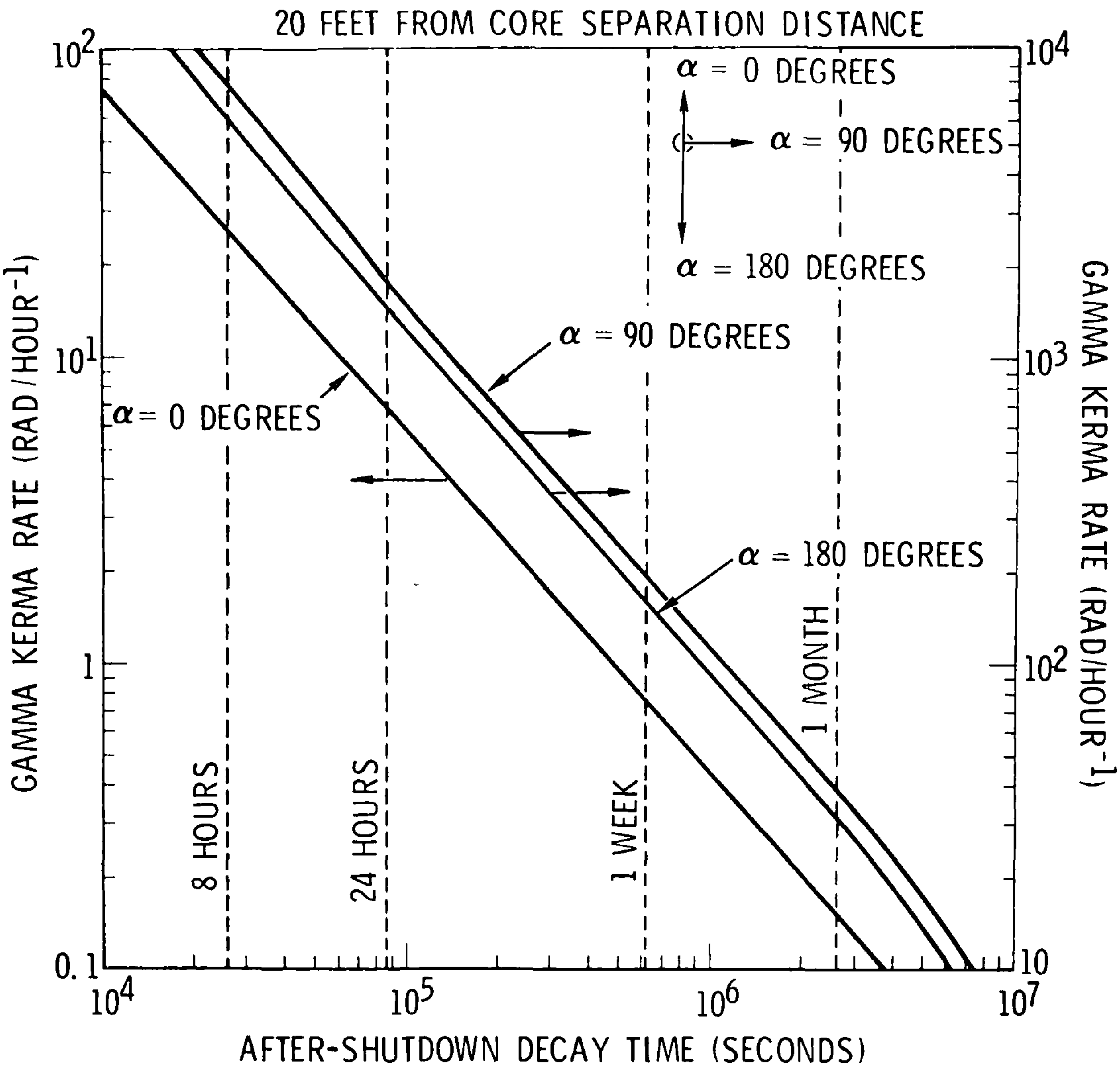

Figure 9-5 After-Shutdown Radiation Environment

5 due to natural decay are evidenced for after-shutdown decay times of 24hours, 1 week, and 1 month, respectively, as illustrated in Figure 9-5 After-shutdown isodose contours 24 hours after the EOI engine diring are shown in Figure 9-6 to further demonstrate a typical after-shutdown radiation environment in the vicinity of the NERVA engine. As in Figure 9-5, gamma kerma rates (rad-hr^{-1}) are given as a functional separation distance in feet from the engine core midplace with reference to a polar angle, $\alpha = 0°$, along the vertical axis in the forward direction.

Figure 9-6 serves to illustrate that only minor stage maintenance operations limited to the 0-15 degree quadrant forward of the engine core midplane appear to be feasible without rather extensive additional shielding provisions up to 24 hours after the last engine shutdown cycle. This observation is predicated on the basic criterion of a 25 rem/year allowable dose to maintenance personnel from the RNS. For example, any operations involving rendezvous maneuvers with a Space Tug for engine removal and/or maintenance will require additional shielding provisions for protection of the crew of the Space Tug, as well as remote handling equipment and manipulative techniques.

It may be concluded that any contemplated mission operations involving a near approach to the engine in either the radial or aft directions will dictate additional fixed shielding requirements.

INTERFACE

The baseline configuration utilizes the INT-21 to launch the RNS tank in an inverted position, whereas the NERVA will be launched by the Space Shuttle as shown in Figure 9.7. Other interfaces include RNS/ground, RNS/payload, RNS/PD and maintenance element.

RNS/Engine Interface

The RNS design criteria related to the RNS/NERVA engine interface is the NERVA Program Requirements Document, SNPO-NPRD-1, revision 9 (dated October 29, 1970) and NERVA reference data (full-flow engine) dated April 1970 (Reference 9.9 and 9.10). Major NERVA engine characteristics which influence the design of the RNS are listed below.

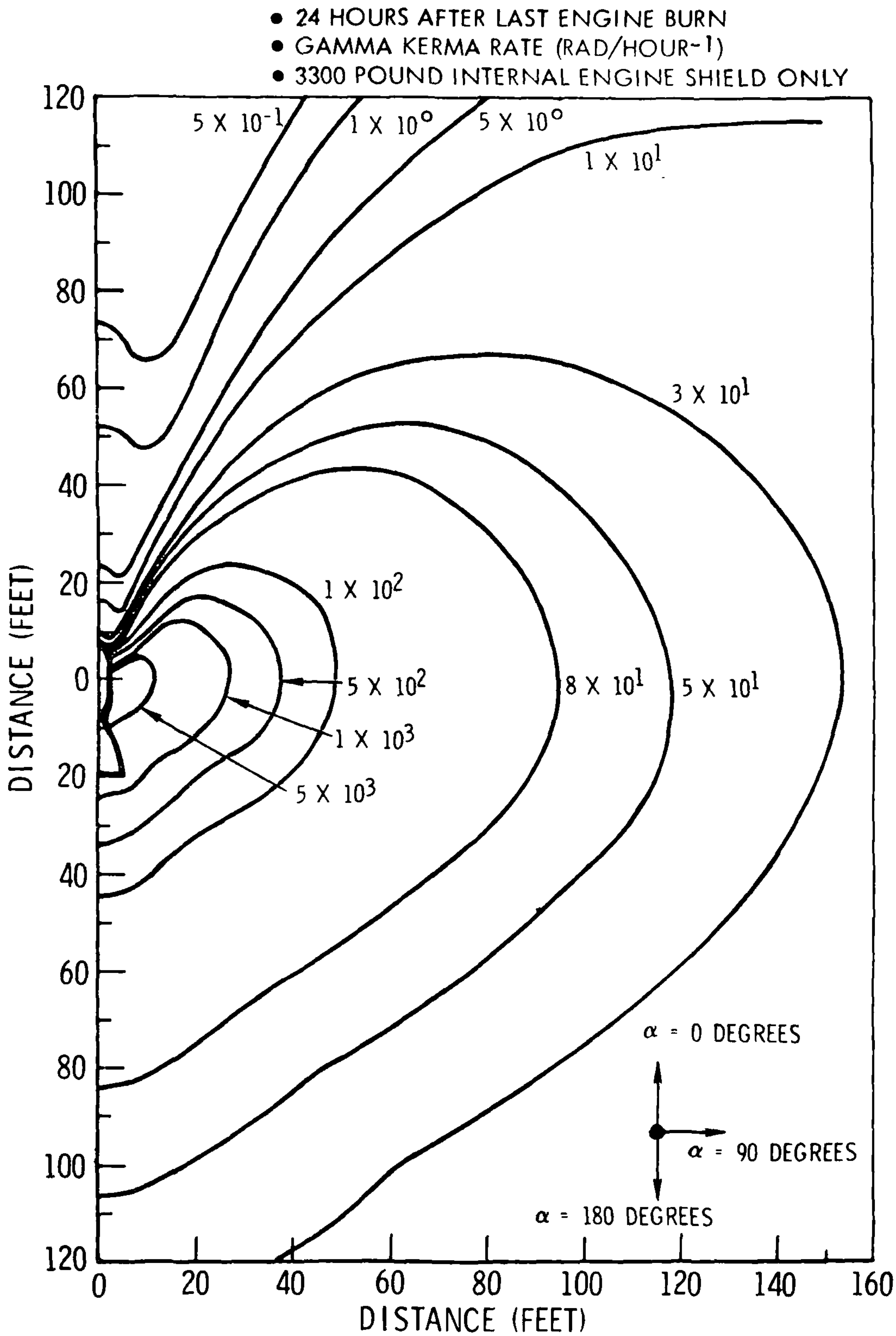

Figure 9 -6 After-Shutdown Iso-Dose Contours

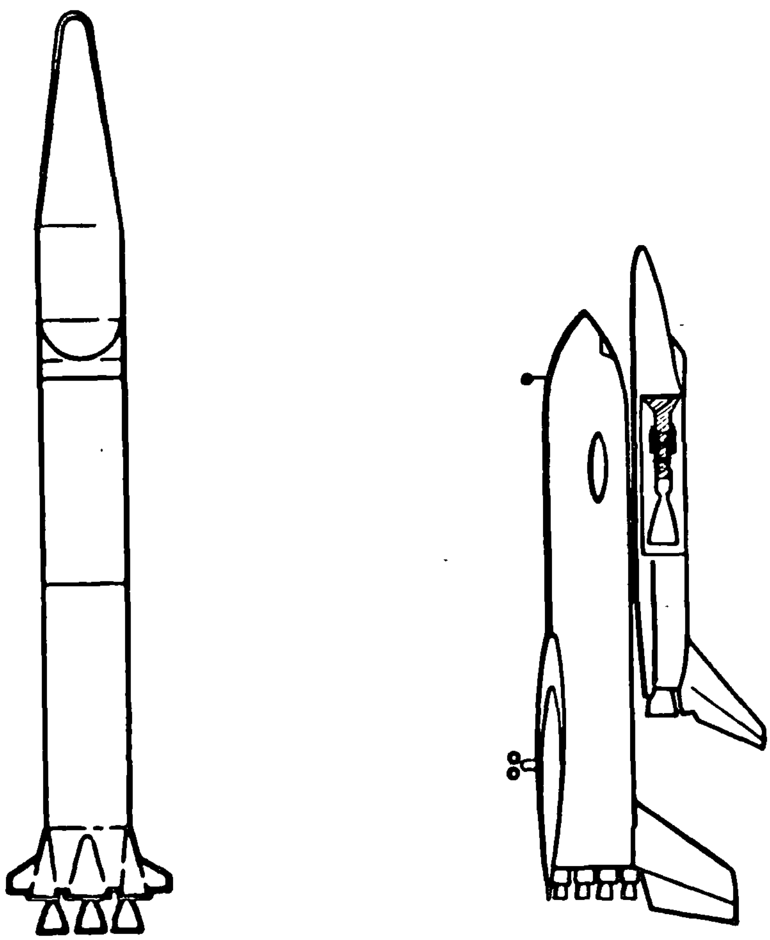

Figure 9-7. RNS Baseline Boost Mode

Performance Requirements at Rated Conditions

Rated conditions shall be based on MSFC Spec. 356A (liquid) hydrogen, delivered at the tank outlet [upstream of the main propellant shutoff valve (PSOV)] at 27.5 psia, and saturation temperature, containing zero percent vapor; with the engine at zero ambient pressure; zero gimbal angle; and single module (engine) operation, utilizing a full-flow cycle to drive the turbopump(s) and assuming a contoured nozzle expansion ratio of 100:1. The engine shall be capable of performing as specified below:

Thrust	75,000±2,000 pounds
Nominal mixed means chamber inlet temperatures	4250°R
Nominal chamber pressure	450 psia
Nominal specific impulse	825 sec
Minimum specific impulse	(TBD) sec
Engine weight maximum	27,800 pounds (including internal shield)
External Shield weight (manned mission)	4050 pounds

The reactor design will incorporate such features as necessary (exclusive of fuel element features) to allow growth, as shown by analysis, to operation of 4500°R nominal mixed mean chamber temperature for two hours duration (12 cycles).

Endurance at Rated Conditions

600 minutes at rated conditions. The operating time shall be utilizable in multiple cycles, up to 60, of varying lengths totaling a minimum of 600 minutes. This rated endurance is not to be construed as the limit of life of all components. Rather, the endurance of such components (except fuel elements) shall be greater than 600 minutes and shall be established by reliability, maintainability, performance, weight, manufacturing, and cost considerations.

A schematic diagram of the full-flow NERVA engine is presented in Reference 9.10. Table 9-1 contains nominal design, steady state points of flow, pressure, and temperature of propellant which was obtained from Reference 9.10. Preliminary RNS/engine interface line sizes are listed below:

Dual turbopump assembly (TPA) feed system (nominal propellant flow rate 91 lb/sec)	9.7 in. dia. each minimum
Stage pressurant supply line	2.25 in. dia.
Coolant supply line	3 in. dia.
Engine purge supply	1 in. dia.

Table 9-1. Nominal State Point Conditions for Typical Normal Performance at Start of Life

State Point Number and Description	Normal Design Point (100% Thrust) (Pc=450, Tc=4250)		
	Flow, lb/sec	Pressure, psia	Temperature, °R
1. Tank Outlet (each)	45.95	30	40
6. Pump Outlet (each)	45.7	1329	59.9
12. Nozzle Manifold Inlet	83.1	1290	60.0
13. Nozzle Outlet	83.1	1122	185.0
20. Dome Baffle Outlet (TIL at Pressure Dome)	91.4	1018	272
25. Turbine Inlet	40.2	984	272
26. Turbine Outlet	40.45	705	247
33. Central Shield Fwd Inlet	91.4	652.7	253
36. Core Inlet	91.4	624	263
37. PV Thrust Chamber Plenum	91.4	450	4250
38. Turbine Bypass Module Inlet	11.0	1001	272
43. Turbine Bypass Module Outlet	11.0	700	272
51. Stem Coolant Line at PV Dome	5.3	1078	60.1
52. Stem Outlet	5.3	1052	896
56. Structural Support Bypass Inlet at PV Dome	3.0	1315	60

Figure 9-8 summarizes the data to be used in RNS performance calculations. The solid curves reflect ANSC data where specific impulse, flow rate, and thrust are averaged over the engine operating phases of startup, steady state, and shutdown. The dashed curves are analytic best fits and extrapolations made to facilitate computer coding and to bridge the region between full power and available data on part power operation. For impulse propellant of less than 1,000 pounds, part power operation is assumed at a specific impulse of 500 seconds and a thrust of 1,000 pounds. Note that for LOI or TEI plane changes the effective I_{sp} will be generally between 500 and 750 seconds.

There is little difference between the use of the reaction control system or the use of the NERVA engine at low power in making midcourse corrections. The NERVA engine will be utilized for making midcourse corrections which will be reflected in RNS operations and designs.

NERVA Engine/Space Shuttle Interface

A mechanical interface between the NERVA engine and Space Shuttle will identify the required propulsion module attach points. Functional requirements will identify conditioning, control, monitor and manipulative requirements associated with boost and orbital assembly.

RNS/Launch Vehicle (INT-21) Interface

The baseline reference point for functional, physical and electrical interface between S-II and RNS will be NASA documents 13M07001, "S-II to S-IVB Stage Physical Requirements," and 40M30593, "Definition of Saturn V Vehicle S-II/S-IVB Electrical Interface." Data in these documents will be utilized as design criteria as applicable to the INT-21 launch vehicle and RNS. ICD's developed during the study for the RNS Launch Vehicle Interface are located in Appendices A and B of volume III of Reference 9.11.

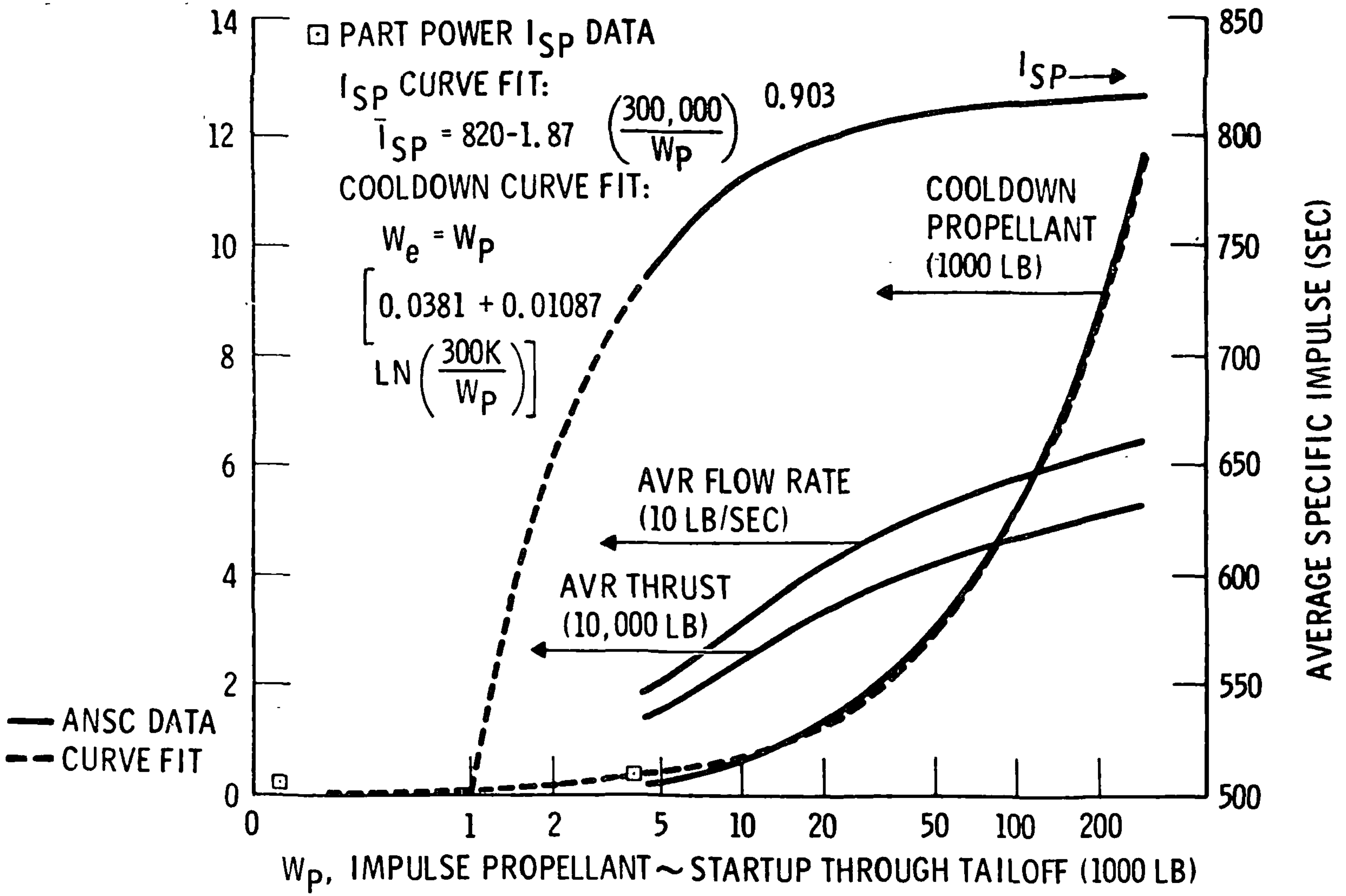

Figure 9-8 NERVA Characteristics for RNS Performance Calculations

Space Division
North American Rockwell

RNS/Ground

Ground support equipment and facilities are designed to meet the requirements of the stage. Ground operations criteria which will influence design of the RNS are:

1. Provisions for handling the stage during fabrication, transportation, test, and assembly on the launch vehicle. Sufficient hard points must be provided to attach handling, servicing, and auxiliary equipment to accomplish the above mentioned operations.

2. Provisions must be made for tank access for inspection and maintenance as well as access to forward and aft areas when mated with the launch vehicle on the launcher transporter.

3. Provisions must be made to permit maintenance of a positive tank pressure after tank closeout, during stage transportation and checkout and/or servicing operation to prevent tank collapse as a result of pressure change resulting from temperature change.

4. Provide for checkout and servicing of RNS subsystems at the fabrication site, acceptance test site, and launch site.

RNS/Payload Interface

No specific payload has been defined by the customer for the RNS. Mission guidelines have identified the types of mission to be flown and the shuttle mission payload weight range but not the payload geometry.

The following guidelines have been established which will have an influence on the RNS/payload and/or PD interface:

1. Refueling of the RNS will be accomplished at the PD under PD control with maintenance being accomplished by maintenance element.

2. The PD provide attitude control and electrical power for the RNS while the RNS is docked to it.

3. The RNS astrionics will be independent of payload. It may provide backup for the INT-21 launch vehicle astrionics unit during Earth launch.

4. For manned operations, man will have override capability for vehicle control.

5. RNS will be checked out prior to each mission.

6. Payloads for the RNS missions will be delivered to and assembled at the Space Station/base.

7. The Space Tug may transfer the payloads between the RNS and the Space Station/base.

8. The RNS will undock from the PD and maintain attitude control while the Space Tug rendezvous and docks the mission payloads.

9. Final automatic checkout of the RNS occurs after vehicle payload docking.

Payloads identified in the Nuclear Flight System Definition, Potential Flight Tests, and Early Operational Payloads Study contract NAS8-24975 have diameters of 260 inches and 396 inches. In-house studies on a lunar orbiting Space Station have identified payloads of 260 inches in diameter. Representative payloads the RNS will deliver to lunar orbit or inject on planetary missions are shown in the section on ICD's of Reference 9.11. A representative advanced manned planetary mission payload is also illustrated there.

FACILITIES

Seal Beach

Overall building heights and building door openings are the major constraints the Seal Beach fabrication site would have on the size of the RNS. Present capability and configurations of the Vertical Assembly Building are shown in Table 9-2 and Figure 9-9. Table 9-3 and Figure 9-10 define the checkout building capabilities. If RNS sizing is such that it exceeds the existing facilities capabilities, this configuration will be used as a point of departure for determining modifications required to accommodate the RNS.

Launch Complex 39

VAB High Bay

The door height of the VAB is 456 feet above ground level. The base platform launch umbilical tower is 25 feet thick. The pedestal height upon which the LUT base platform rests is 22 feet. The crawler transporter which transports the LUT and launch vehicle from the VAB to the launch pad has an adjustable height of 20 to 26 feet. With the LUT resting on the pedestal, the

Table 9-2. Vertical Assembly Hydrotest Building

Item	Description
BRIDGE CRANES	
Station I	2-ton bridge with one 2-ton hoist with pushbutton control from each floor. Hook height 107 ft 3 in. 20-ton bridge with two 10-ton hoists, cab operated, hook height 103 ft 3 in.
Station II	20-ton bridge with two 10-ton hooks, cab operated, hook height 104 ft 11 in., 5-ton bridge with one hook, cab-operated, hook height 104 ft (same crane used in Sta. IV).
Station III	20-ton bridge with two 10-ton hoists, hook height 103 ft 3 in. (same crane used in Station I).
Station IV	20-ton bridge with two 10-ton hooks, hook height 104 ft 11 in. (same crane used in Station II).
Station V	Two 70-ton bridge with two 35-ton fixed hooks cab-operated, 110-ft hook height with one 5-ton auxiliary hoist, 110-ft hook height each bridge.
Station VI	One 50-ton bridge with two 25-ton fixed hooks cab-operated, hook height 52 ft 2 in. with one 5-ton auxiliary hoist, hook height 52 ft 2 in. One 10-ton bridge with one 10-ton hoist.
Station IV Roof	One 5-ton bridge crane, pendant operated with one control station at roof level and one at ground level. Hook height 145 ft Vertical Tank Entry.

Table 9-2. Vertical Assembly Hydrotest Building (Con't)

Item	Description
PLATFORMS	
Station I	Fixed platforms and hinged platforms at all levels - 16, 32, 48, 64, and 80 feet.
Station II	Fixed platforms and hinged platforms at all levels - 16, 32, 48, 64, and 80 feet.
Station III	Fixed platforms at 16,32,48,64, and 80 ft. Hinged platforms at 16, 32, 48, 64, and 80 ft (as of 3-66).
Station IV	Fixed platforms at 16, 32, 48, 64, and 80 ft. Hinged platforms at 16, 32, 48, 64, and 80 ft.
Station V	Fixed platforms at 16, 32, 48, 64, and 80 ft. One moveable platform on front side of station that travels from approximately 8 feet to 80 feet.
Station VI	Fixed platform around 36-ft diameter opening at 16-ft level.
LARGE DOOR OPENINGS	
South of Stations I, II, III, and IV	100 ft high, 47 ft 6 in. wide.
Station V	119 ft high, 43 ft 6 in. wide, includes guillotine opening.
FLOOR LOADINGS	Concentrated floor load of 2000 lb on any 4 sq-ft area with an average of 60 lb per sq ft.
SUBSTATIONS	2 at 1500/1725 kva; 12 kv to 480/277 volts
AREA	109,280 sq ft (gross)
BLDG. DIM.	170 x 270 ft
BLDG. HEIGHT	120 feet

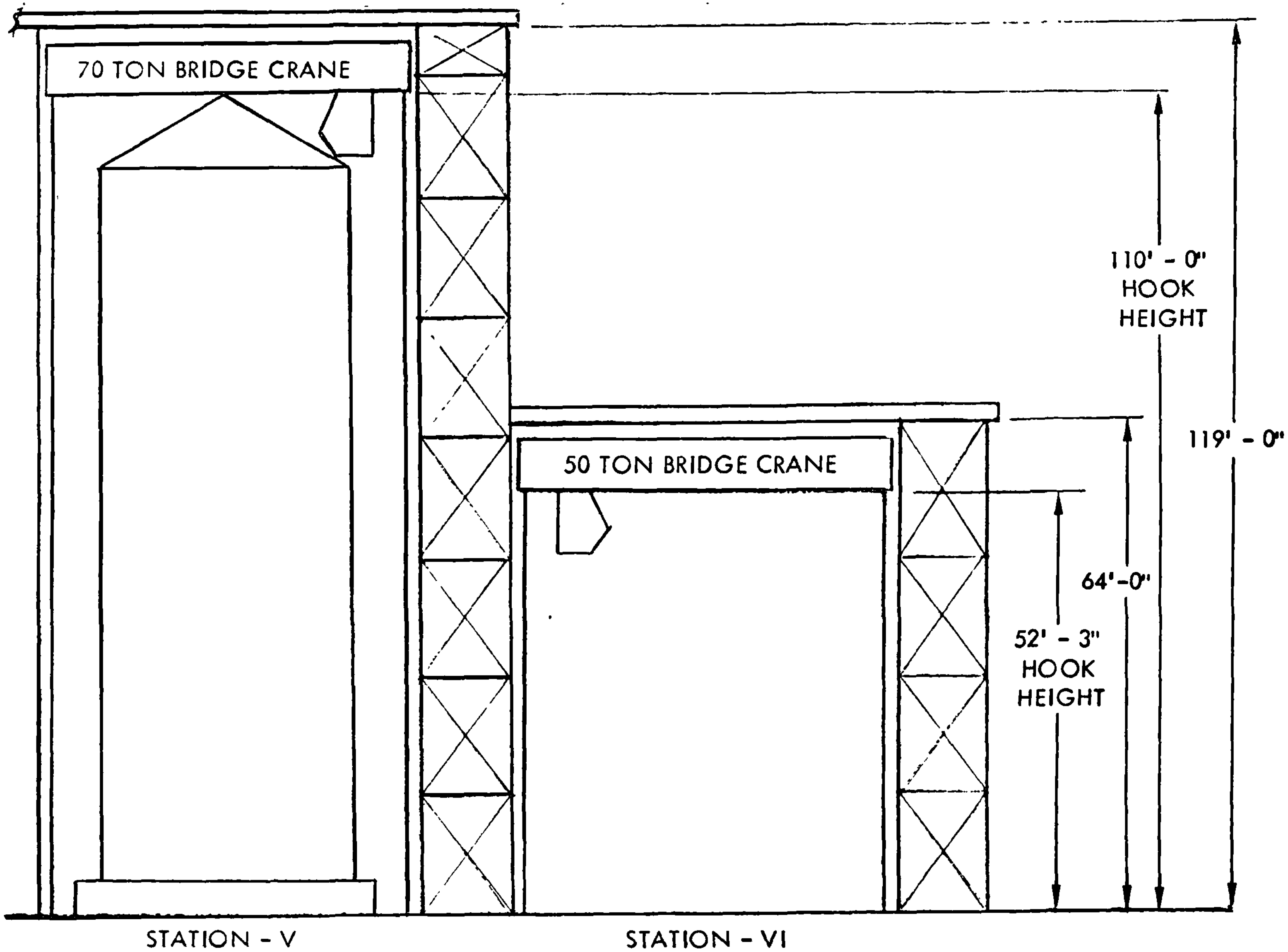

Figure 9-9 Vertical Assembly Building

Table 9-3. Vertical Checkout Building

Item	Description
BRIDGE CRANES	None
STIFF LEG DERRICKS	
1	50-ton
1	70-ton
LARGE DOOR OPENINGS	
Station VIII	North door 45 ft wide x 108 ft high
Station IX	North door 45 ft wide x 108 ft high
Station VIII	South door 13 ft 10 in. wide x 20 ft 3 in. high
Station IX	South door 13 ft 10 in. wide x 20 ft 3 in. high
FLOOR LOADINGS	
Plenum Floor	Second floor control room - 200 lb/sq ft
Stage Area	150 lb/sq ft
SUBSTATIONS	1500 kva 12 kv to 480/277 volts
AREA	51,940 sq ft (gross)
BLDG. DIM.	
Vertical Sta.	56 x 132 ft
Block House	44 x 160 ft
BLDG. HEIGHT	135 ft

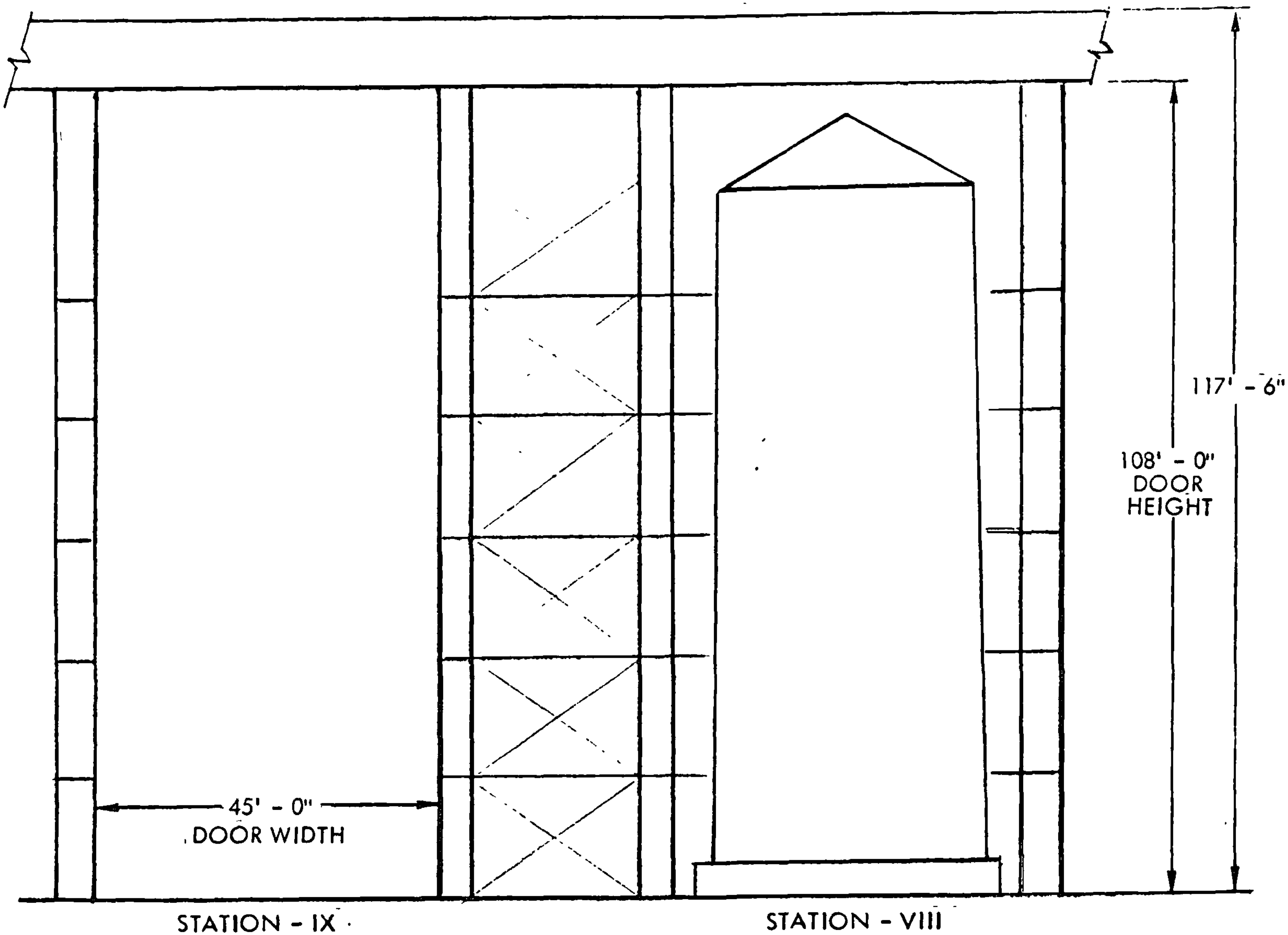

Figure 9-10 Checkout Building

zero level reference point is 47 feet above the ground level. Assuming the crawler transporter must be extended to the full 26-foot height to clear the pedestal, the maximum vehicle height is constrained to 405 feet above the zero reference plane. The S-IC stage engines of the Saturn V launch vehicle extend 115.5 inches below the zero reference plane, therefore, the overall vehicle height may reach 414 feet.

The maximum height to which the high bay hook can be raised is 462 feet. This hook height should be taken into consideration when VAB hoisting operations are required and handling equipment is designed for the high bay assembly operations.

Height of the INT-21 launch vehicle above station zero reference is 2519 inches or approximately 210 feet. The distance from the INT-21 interface to the door obstruction is 195 feet with the crawler transporter extended to its maximum height of 26 feet to clear the door obstruction which would result in an INT-21 payload shorter than 195 feet.

Figure 9-11 depicts the VAB high bay dimensional constraints discussed above.

A 175-ton overhead bridge crane serves the transfer aisle between the high bay and low bay areas. Each pair of high bays is served by a 250-ton bridge crane with a hook height of 462 feet.

Low Bay

Transfer aisle door opening is 55 feet wide by 94 feet high. The 175-ton bridge crane that serves the transfer aisle has a hook height of 166 feet. The low bay contains eight stage preparation and checkout cells. Five-ton monorail hoist hook height in preparation and checkout cells is typically 110 feet 6 inches. Test cell ceiling height is 116 feet. Reference 9.12 is the source data for the VAB constraints discussed above.

Table 9-4 presents existing VAB facility and GSE limitations that will influence the RNS configuration that can be accommodated at each of the possible operational locations. The high bay door clearance of 456 feet is the main limitation to the length of an RNS that can be assembled in the VAB. The length of the RNS is limited by the height of the mobile launcher (47 feet), the S-IC stage (138 feet), and S-II stage (81.5 feet) for a total height of 265.5 feet less 9.5 feet S-IC engines which extend below the ML platform, or a total stack height of $\approx$ 256 feet. Based on the assumption that ten feet of clearance is necessary to roll out the ML with the INT-21/RNS, establishes a total RNS length of 190 feet (including the nose cone of $\approx$ 25 feet). Assuming it is possible to mate the nose cone outside the VAB (by incorporating a crane over the high bay door), the RNS length that can be handled will increase to 215 feet outside the VAB or 190 feet (without nose cone) within the VAB.

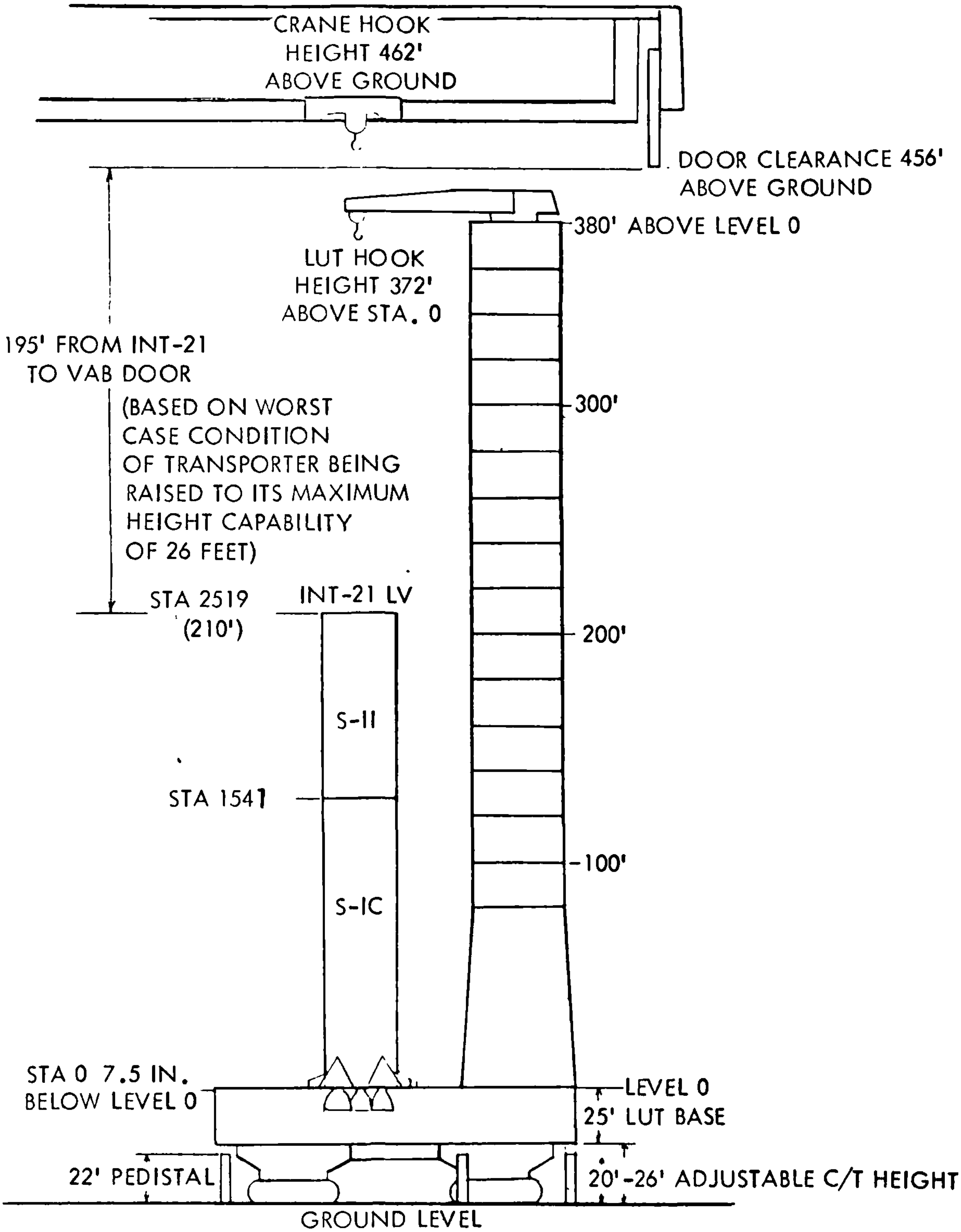

Figure 9-11 Launch Complex 39 VAB High Bay Clearance

Table 9-4. VAB Facility Limitations

MAJOR ITEMS	LOW BAY AISLE	LOW BAY TEST CELL	HIGH BAY
TRANSFER AISLE (FT)			
HEIGHT	210		525
WIDTH	92		92
LENGTH	715		715
BRIDGE CRANE (TONS)	175	5	250
CRANE HOOK HEIGHT (FT)	166	114	462
MAXIMUM DOOR CLEARANCE (FT)	94	116	456
CRAWLER TRANSPORTER HEIGHT (FT)			47
MAXIMUM INT-21/RNS STACKED (FT)			409
S-IC/S-II STACKED ON MOBILE LAUNCHER (FT)			209
MAXIMUM RNS CONFIGURATION LENGTH (FT)	160	100	190

In all cases of operational concepts, whether the engine-tank-interstage are mated horizontally or vertically, with one piece or segmented interstage, the maximum RNS vehicle that can be accommodated in the VAB is limited to 190 feet.

The above limitations apply only to stage configurations requiring main propellant tank/engine/interstage mating on the ground. For those configurations employing orbital assembly of the operational system, the two dimensional limits are 160 and 190 feet if assembly and checkout operations are conducted in the low bay aisle and high bay, respectively. This is approximately 35 feet longer than launching an assembled engine-stage-interstage. It is to be noted that in order to get maximum benefit of a stage length when stage is launched upside down, the nose cone has to be mated to the stage before stacking the stage on top of INT-21 booster.

FABRICATION AND MATERIALS CONSTRAINTS

Fabrication Imposed Constraints

(to be determined)

Material Constraints Imposed by Radiation

The radiation damage effects on structural metals and alloys, organic materials (plastics, adhesives, insulation, laminates, etc.), electronic devices, semiconductor devices, and materials must be taken into consideration for RNS designs.

A preliminary radiation damage assessment of possible nuclear stage materials, components and subsystems was conducted in the Phase II study and data generated are documented in Volume 3 of Reference 9.11.

TRANSPORTATION AND STORAGE

Transportation modes for RNS are essentially that utilized for the Saturn S-II stage. The transporter pallet can be modified to accommodate the RNS length. Overland transportation will be provided by the Type II stage transporter and an M-26 prime mover.

Water transportation will be provided by the AKD utilized in transporting the Saturn S-II stage. Tiedown modifications will be required to accommodate the length of the RNS.

Storage - TBD.

SAFETY

Recommended Nuclear Flight Stage safety criteria has been documented in Reference 9.13. This analysis took into account the orbital start modes of operation. Since orbital start is the mode of operation for the RNS, only that design criteria applicable to orbital start is identified for RNS design.

The implications of the safety criteria for the RNS design effort are given as follows in the form of broad design requirements:

1. Central poison wires will be removed after initial EOS rendezvous with the RNS. The system which is responsible for removing the (ACPWS) poison wires will be designed to avoid core or nozzle injury during removal.

2. All crucial subsystems shall be designed to be "failsafe," i.e., in the event of a component or electrical system failure, the reactor will automatically proceed to a constant power mode or shutdown mode as required by the mission.

3. The stage nuclear engine control system shall be sufficiently independent and isolated from all of the circuits or energy sources to prevent credible inadvertent start or loss of control due to failures in other systems or to spurious electrical signals.

4. All nuclear engine control systems must provide parallel and independent signal paths from separate power sources.

5. Diagnostic instrumentation shall be provided to detect incipient failures of types identified as crucial.

6. Select materials to reduce probability of fire, toxicity, corrosion and electrical shock hazard.

Safety related design criteria and constraints identified in NASA's guidelines document are listed below.

1. RNS will be man-rated.

2. All credible single failure modes or credible combinations of failures and errors which result in loss of crew and passengers or unacceptable risk to generate population groups will be eliminated by design change and/or mission modification.

3. No single failure or credible combination of failures and errors will prevent or preclude operation of the NERVA engine in the emergency mode

4. Total radiation dose from the NERVA engine and plume sources will be limited to 10 rem per round trip shuttle mission at the tank top.

5. RNS maintenance personnel will not receive more than 25 rem per year from RNS.

6. Total integrated radiation dose from the RNS to any manned Space Station or manned orbital system will not exceed 0.1 rem during any single NERVA engine burn.

Emergency detection and operation of the NERVA engine relating to item 3 above are identified in the following paragraphs.

Maximum effort shall be placed on a design which eliminates single failures or credible combinations of errors and/or failures which endanger the completion of the mission, the flight crew, the launch crew or the general public.

In the event the planned mission must be abandoned the effect of each mode of failure on engine performance must be minimized so as to make the optimum use of remaining propellant and at the minimum to provide an integrated engine performance of·

(1) 30,000 pounds thrust
(2) 500 seconds specific impulse
(3) 10^8 pound-seconds total impulse including cooldown propellant with the total impulse controllable up to a maximum of 10^8 pound-seconds.

These requirements shall be provided by a single engine cycle in the degraded state. Operation in an emergency mode must be attainable from all operating modes of the engine cycle including all shutdown modes and coast phases. If mission abandonment is required during the normal steady state operation immediate retreat to an emergency mode shall be made. If the malfunction occurs during engine modes following the steady state power modes, provision shall be made for cooling up to five hours prior to entering an emergency mode. Ramps from the normal operational modes to an emergency mode are TBD. Shutdown ramps from the emergency mode are TBD. Final cooling shall preclude engine disassembly and, if it can be done at no additional risk to population, passengers or crew, preserve the engine in a restartable condition.

The required total impulse must be provided in a sustained emergency mode at any of the points above the required minimum and within the normal operating map. The operating point selected after the failure has occurred will be determined by the nature of the failure and the reliability of retreating to and operating at the emergency mode operating point.

Potential failure modes that would preclude attainment of these requirements shall be identified and presented to the Government for review with justification for retention.

A contingency Analysis and Planning Process shall be continuously conducted to ensure that these requirements are met.

In addition, the NERVA engine shall incorporate the following features:

(1) Means of preventing accidental criticality during all ground and space operations. An anticriticality destruct system shall be provided for launch and ascent when launched by an unmanned vehicle.

(2) Throughout the mission means shall be provided for preventing credible core vaporization, disintegration or violation of the thrust/load path to the payload.

(3) Judicious selection of diagnostic instrumentation adequate to detect the approach of a failure or an event which could injure the crew or damage the spacecraft directly and provisions to preclude such an event.

(4) Judicious selection of diagnostic instrumentation adequate to detect deteriorating situations or incipient failures.

(5) Ability to override the engine programmer remotely by the crew and ground control, the ability for remote thrust cutdown independent of the engine program. In addition, the engine control system shall have the ability to preclude excessive or damaging deviations from programmed power and ramp rates.

STRUCTURAL DESIGN CRITERIA

General Requirements

The contractor shall show by analyses that the preliminary structural design meets the design requirements with sufficient margin of safety to assure adequate strength, rigidity, and safety of personnel at all times. The stage shall be designed to minimize weight and yet resist all loads and combination of loads that may reasonably be expected to occur during all phases of fabrication, testing, transportation, erection, checkout, launch, flight, and recovery. The design criteria shall be furnished by MSFC (Reference 9.14) Criteria originated by the contractor may be used after obtaining MSFC approval. Methods of analysis, material allowables, and formulas shall be adequately referenced to MIL-HDBK-5 or other supplemental documents as may be determined by MSFC to be acceptable standard references. When used in addition to MIL-HDBK-5, supplemental documents shall be in a published form and available to MSFC upon request. The computer programs utilized must be documented and approved by MSFC. The material strength minimum guaranteed values shall be used. In selecting material strength allowables from materials that show no pronounced yield point, the yield point shall be the 0.2 percent offset value. Material strength allowables shall include all environmental effects to which the material will be exposed from fabrication through flight. For materials where the yield point cannot be established, the safety factor against ultimate shall govern.

Definitions

The following definitions and terms shall be used for design and analysis of the stage or vehicle and in all documentation to establish uniform nomenclature with respect to loads, safety factors, etc. The factors of safety to be used in the design and analysis of the structural and propulsion systems are also listed in this standard.

Limit Load - Limit load is the maximum load calculated to be experienced by the structure under the specified conditions of operation and includes steady stage load, using the appropriate acceleration, and dynamic load due to environment or forcing function.

Design Load - The design load is the limit load multiplied by the required minimum factor of safety.

Allowable Load - (Stress) - The allowable load or stress is the maximum load or stress a particular element can be subjected to, as for example· if buckling is the design criteria, then the allowable load on a column is the load which causes buckling.

Factor of Safety - The factor of safety is defined as the ratio of the allowable load or stress to the limit load or stress.

Margin of Safety - The margin of safety is the percentage by which the allowable load or stress exceeds the design load or stress. For example, allowable stress can mean the material yield stress, the material ultimate stress, etc.

Operating Pressure - The operating pressure is the nominal pressure to which the components are subjected under steady stage conditions in service operations.

Limit Pressure - The limit pressure is the maximum operating pressure or the operating pressure including the effect of system environment such as vehicle acceleration. For hydraulic and pneumatic components and assemblies, limit pressure will include the effect of pressure transients.

Proof Pressure - The pressure to which every production pressure containing component is subjected. Unless otherwise approved by the procuring agency the proof pressure shall be run at operating temperatures. Every point in the test article shall be subjected to at least the proof factors specified in Factor of Safety section below. Where practical the proof pressures shall also be used to envelope the combined effect of external load and internal pressure. The component shall not exhibit any deformation detrimental to operation of the component at the proof pressure level.

Yield Pressure - A test level as specified in Factor of Safety section below to which the qualification article of certain pressure containing components is subjected. Unless otherwise approved by the procuring agency the yeidl pressure test shall be run at operating temperature. The component shall not exhibit permanent deformations other than restrained local yielding pressure.

Ultimate Pressure - A test level as specified in Factor of Safety section below to which at least one qualification article of every pressure containing component is subjected. Unless otherwise approved by the procuring agency the ultimate pressure test shall be run at operating temperature. The component shall not rupture or leak at the ultimate pressure level.

Combined Stresses - Combined stresses are stresses resulting from the simultaneous action of all loads and environments.

<u>Factor of Safety</u>

The following factors of safety are the minimum to be applied. These factors shall be applied to the combined stresses with the following exception:

In circumstances where certain loads have a relieving, stabilizing, or otherwise beneficial effect on structural load capability, the minimum expected value of such loads shall be used and shall not be multiplied by the factor of safety in calculating the design yield or ultimate load. For example, the ultimate compressive load in pressurized vehicle tankage shall be calculated as follows:

Ultimate Load = Safety Factor x Body Loads - Minimum Expected Pressure Load

For components or systems subjected to several missions, safety factor requirements shall apply to the final mission.

(1) General Safety Factors

Manned Vehicle	
Yield Factor of Safety	= 1.10
Ultimate Factor of Safety	= 1.40
Unmanned Vehicle	
Yield Factor of Safety	= 1.10
Ultimate Factor of Safety	= 1.25

(2) Propellant Tanks

Manned Vehicle	
Proof Pressure	= 1.05 x limit pressure
Yield Pressure	= 1.10
Ultimate Pressure	= 1.40
Unmanned Vehicle	
Proof Pressure	= 1.05 x limit pressure
Yield Pressure	= 1.10
Ultimate Pressure	= 1.25

(3) Hydraulic or pneumatic systems

Flexible hose, tubing and fittings less than 1.5 inch in diameter	
Proof pressure	2.00 x limit pressure
Burst pressure	4.00 x limit pressure
Flexible hose, tubing and fittings (including LOX and LH_2 vent lines) 1.5 inch in diameter and greater	
Proof pressure	1.50 x limit pressure
Burst pressure	2.50 x limit pressure
Gas reservoirs	
Proof pressure	1.50 x limit pressure
Yield pressure	1.10 x present pressure
Burst pressure	2.00 x limit pressure

(4) Actuating cylinders, valves, filters, switches

Proof pressure	1.50 x limit pressure
Burst pressure	2.50 x limit pressure

(5) LH_2 feed lines

Proof factor of safety	1.05
Yield factor of safety	1.10
Burst factor of safety	1.40

Note: The factors of safety of (3), (4), and (5) are never used in combination with those shown under (1) and (2).

(6) Solid Motor Casings

Proof Pressure	= 1.05 x limit pressure
Yield Pressure	= 1.20 x limit pressure

When a pressurized system or components subjected to external loads, such as air loads, ground handling, transportation, in addition to pressure, factors of safety given above will be used. That is, the pressure vessel thickness is determined by the use of applicable pressure factors and then the component is analyzed for the external loads pressures, and environments with the general safety factor.

When adequate fracture roughness data and sufficient knowledge of operating conditions are available to determine the required proof pressure from fracture mechanics principles, the required proof pressure may be determined from this data and used instead of the safety factors listed above. Written approval by MSFC will be required.

Fatigue Analysis

All structural elements shall be evaluated for their capability to sustain any cyclic load condition which is part of the design environment. For those elements whose design is controlled by a cyclic or repeated load condition, or a randomly varying load condition, a preliminary fatigue analysis will be conducted if sufficient knowledge of the operating conditions are available.

If sufficient fatigue data is available to establish statistical minimum guaranteed fatigue allowables, the component shall be capable of withstanding three times the predicted number of load cycles. If only typical fatigue allowables are available the component shall be capable of withstanding ten times the predicted number of load cycles. For cyclic loads to varying levels such standard methods as Miner's method shall be used to determine the combined damage. For repeated load combined with a steady load such standard methods as the modified Goodman diagram shall be used to determine the combined effect.

Handling and Transportation Factors

As a design goal, flight structures shall not be subjected to transportation and handling loads more severe than flight design conditions.

The transportation loads are a function of the transportation mode. The transportation loads shall include the steady state loads plus dynamic, vibration, and shock loads which have to be determined based on the mode of transportation by analysis.

The transportation equipment design loads shall contain a factor sufficient to assure that no structural problems will arise to jeopardize the flight hardware.

REFERENCES

9.1 MSFC Document No. PD-SA-P-70-63, "Guidelines and Constraints Document Nuclear Shuttle Systems Definition Study Phase A," Revision No. 2, dated May 28, 1970

9.2 MSFC Document No. PD-SA-25-69(OT) Nuclear Vehicle Operating Requirements

9.3 Boeing Report D5-15583

9.4 SNPO-NPRD-1, NERVA Program Requirements Document, Release 7, dated January 19, 1970

9.5 NASA TMX-53872, Terrestrial Environment (Climatic) Criteria Guidelines for Use in Space Vehicle Development, 1969 Revision (second printing)

9.6 NASA TMX-53957, Space Environment Criteria Guidelines for Use in Space Vehicle Development (1969 Revision)

9.7 NASA TMX -53865, Natural Environment Criteria for the NASA Space Station, (second printing)

9.8 MPR-SAT-FE-69-1, Saturn V Launch Vehicle Flight Evaluation Report-AS-503-Apollo 8 Mission, dated February 20, 1967

9.9 SNPO-NPRD-1, NERVA Program Requirements Document, Revision 9, dated October 29, 1970

9.10 ANSC Report S-130-CP09029-AF1, "NERVA Reference Data NERVA Program," Contract SNP-1, dated September 1970

9.11 SD70-117, Nuclear Flight System Definition Study Phase II Final Report, August 1970

9.12 Apollo/Saturn V Facility Description (K-V-012) Volume II of IV Volumes Launch Complex 39 Facility Description, October 1966

9.13 SD69-498, "FY 1969 IR&D Report Nuclear Flight Stage Safety Analysis," dated October 1969

9.14 PD-SA -P-70-193, Guidelines for Structural Design Criteria, August 7, 1970 (S&E-ASTN-AS-70-55)